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ALTITUDE TESTING OF THE J-2 ROCKET ENGINE IN PROPULSION ENGINE TEST CELL (J-4) (TESTS J4-1554-01 THROUGH J4-1554-11)

W. W. Muse and D. E. Franklin

ARO, Inc.

July 1967

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LARGE ROCKET FACILITY
ARNOLD ENGINEERING DEVELOPMENT CENTER
AIR FORCE SYSTEMS COMMAND
ARNOLD AIR FORCE STATION, TENNESSEE

NOTICES

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FOREWORD

The work reported herein was sponsored by the National Aeronautics and Space Administration (NASA), Marshall Space Flight Center (MSFC), under System 921E, Project 9194.

The results of the tests presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), Air Force Systems Command (AFSC), Arnold Air Force Station, Tennessee, under Contract AF40(600)-1200. Program direction was provided by NASA/MSFC; engineering liaison was provided by North American Aviation, Inc., Rocketdyne Division, the responsible agency for the J-2 engine, and Douglas Aircraft Company, the responsible agency for the S-IVB stage. Testing was conducted during the period from July 30 to November 16, 1966, in Propulsion Engine Test Cell (J-4) of the Large Rocket Facility (LRF) under ARO Project No. KA1554. The manuscript was submitted for publication on April 7, 1967.

This technical report has been reviewed and is approved.

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ABSTRACT

Eleven test periods involving a total of 14 J-2 rocket engine starts were accomplished in Propulsion Engine Test Cell (J-4) to verify the engine altitude ignition characteristics and performance in support of the Saturn IB and Saturn V flights. The test article consisted of a flight configuration J-2 rocket engine and a "battleship" S-IVB stage. Engine ignition characteristics were different from those predicted for altitude testing and provided an important insight into the J-2 engine transient operation for the Saturn IB and Saturn V flights. Transient gas generator outlet temperatures observed were consistently higher than engine acceptance test values. One test resulted in an engine safety cutoff because of an excessive transient gas generator outlet temperature. Combustion instability during engine ignition was observed for time durations ranging from 3 to 132 msec for ten of the firings and was similar to the instability recorded during the Saturn IB AS-201 through -203 flights. Excessive engine vibration resulted in an engine safety cutoff on one test. Engine side forces measured at altitude conditions were significantly less than those recorded during previous J-2 engine tests. Engine operation time accumulated during these tests was 145.8 sec.

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NOMENCLATURE

A	Area, in.^{2}
ASI	Augmented spark igniter
ES	Engine start, designated as the time that He control and ignition phase solenoids are energized
ESCS	Engine safety cutoff system
GG	Gas generator
GGOT	Gas generator outlet temperature
GSE	Ground support equipment
MOV	Main oxidizer valve
O/F	Oxidizer-to-fuel propellant mixture ratio
PU	Propellant utilization
STDV	Start tank discharge valve
t_o	Defined as the time at which an opening signal is applied to the STDV solenoid
VSC	Vibration safety count, an indicator of the time duration of engine vibration in excess of 150 g

SUBSCRIPTS

e	Exit
f	Force
fc	Full closed
fo	Full open
fp	Fuel poppet
ic	Initial closing
io	Initial opening
m	Mass
op	Oxidizer poppet
t	Throat

SECTION I
INTRODUCTION

The J-2 rocket engine was designed and developed under contract to the National Aeronautics and Space Administration (NASA) by North American Aviation, Inc., Rocketdyne Division, for the Saturn IB and Saturn V launch vehicles (Fig. 1) in support of the NASA Apollo Program to explore the surface of the moon and for subsequent outer space missions. The J-2 (Fig. 2) is a 200,000-lb_f-thrust class, multiple restart engine utilizing LO_2 and LH_2 as propellants.

During early June, 1965, preliminary discussions were held between representatives of AEDC and NASA regarding the feasibility of conducting tests with the J-2 rocket engine in its S-IVB stage configuration in the Propulsion Engine Test Cell (J-4) of the Large Rocket Facility (LRF) (Fig. 3, Ref. 1). A need for static testing of the J-2 engine at altitude conditions to verify the engine ignition characteristics and performance in the range of conditions expected on the early Saturn flights was recognized by NASA. Altitude testing at AEDC would allow the J-2 engine to be subjected to conditions near the expected space environment and would provide a facility to investigate engine difficulties in the event that any were encountered in the altitude verification tests or in the early Saturn flights. An environmental verification program at AEDC was desired because the program cost would be only a fraction of the amount required to evaluate the integrity of the J-2 engine during a Saturn vehicle flight. On August 27, 1965, NASA authorized the J-2 Engine Environmental Verification Test (EVT) Program to proceed. The objectives of the test program were originally established as follows:

1. Primary Objectives
   - a. Simulate engine thermal conditions for both single burn and restart applications of the J-2 engine and evaluate the start transient characteristics at a pressure altitude of approximately 100,000 ft.
   - b. Evaluate thrust chamber side-force characteristics and engine performance.
   - c. Evaluate thrust chamber temperatures at engine start (ES) and the He usage for the simulated boost phase and orbital coast conditions.
2. Secondary Objectives

   Acquire data to update the J-2 engine start and shutdown conditions and to better define the engine operating limits.

Certain facility modifications were necessary for the conduct of the test program, the most significant being the addition of a testing capability for LH_2-fueled engines. The S-IVB "battleship" stage arrived at AEDC on February 22, 1966; the J-2 engine arrived on March 4, 1966. The installation of the test article began on April 16, 1966.

Eleven test periods involving a total of 14 ES and 145.8 sec of engine operation were accomplished at pressure altitude conditions from July 30 to November 16, 1966. The original program commitments called for the initial test to be conducted in July 1966. In view of unpredicted start transient characteristics on the early tests, additional emphasis was placed on the investigation of engine transient characteristics. As a result, the test objectives for the 11 tests reported herein were revised as follows:

1. Evaluate the engine transient operation and performance at a pressure altitude of approximately 100,000 ft.
2. Acquire data to better define the engine operating conditions and limits for flight applications.

SECTION II
APPARATUS

2.1 TEST ARTICLE

The test article was an S-IVB stage of the Saturn IB and Saturn V vehicles and was comprised of the J-2 rocket engine (S/N 2052), designed and developed by Rocketdyne, a division of North American Aviation, Inc., and an S-IVB battleship stage, designed and developed by the Douglas Aircraft Company. The integrated J-2 engine and S-IVB flight stage assembly is presented in Fig. 4.

The J-2 rocket engine is a multiple-restart engine that utilizes LO_2 and LH_2 as propellants and is designed to be used singly or in clusters. Thrust rating of the engine is 225,000 lb_f at an oxidizer-to-fuel mixture ratio (O/F) of 5.5.

The boilerplate-constructed S-IVB stage provides propellant capacity for testing the engine in the flight configuration. The fluid dynamics of the stage are identical to the flight vehicle. The weight of the battleship stage (excluding propellants) is approximately 60 tons; the weight of a flight stage is approximately 11 tons.

Twenty-seven modifications were made to the engine after acceptance testing and before shipment to AEDC. A modification having a decided effect on the engine transient operation was the gas generator (GG) control system orifice modification. Engine modifications and unsatisfactory component replacements performed at AEDC are presented in Tables I and II, respectively.

2.1.1 J-2 Rocket Engine

The J-2 rocket engine (Fig. 5, Ref. 2) features a bell-shaped thrust chamber, an injector, two direct-drive turbopumps, a gas generator, engine valves, engine-mounted electrical and pneumatic controls, and instrumentation. Initial fuel and oxidizer turbopump start impulse is delivered by an engine-mounted start tank which is pressurized with GH_2. The nominal engine O/F ratio ranges from 4.5 to 5.5, depending upon the propellant utilization (PU) valve position. A pneumatic control system is used for engine valve operation and obtains its energy from regulated GHe supplied from an engine-mounted tank. Engine-mounted electrical control packages provide the necessary logic for proper engine sequencing during operation.

The tubular-wall, bell-shaped thrust chamber (Fig. 6) consists of a cylindrical combustion chamber, a 170.4-in.^{2} throat area, and a divergent nozzle with an expansion ratio (A_e/A_t) of 27.1. The 18.6-in.-diam combustion chamber is 8.0 in. long from the injector mounting face to the throat inlet, and the characteristic length (L^*) is 24.6 in. Overall thrust chamber length is 133 in. from the fuel pump low pressure duct inlet to the nozzle exit. The thrust chamber body is constructed of longitudinal stainless steel tubes that are brazed together and are supported by external stiffening bands around the tubes. Thrust chamber cooling is accomplished by fuel flow, which is supplied to a fuel manifold, circulated downward through 180 tubes, and circulated upward through 360 tubes to the thrust chamber injector.

The thrust chamber injector (Fig. 7) is a concentric-orificed (concentric fuel orifices around the oxidizer post orifices), porous-faced injector. Fuel and oxidizer injector orifice areas are 25.0 and 16.0 in.^{2}, respectively. The porous material, forming the injector face, allows three or four percent of fuel to flow through and cool the face of the injector.

The augmented spark igniter (ASI) unit is mounted on the thrust chamber injector (Fig. 8) and supplies the initial energy source to ignite propellants in the main combustion chamber. The ASI chamber is an integral part of the injector (Fig. 7). Fuel and oxidizer are routed to the ASI combustion area, where they are ignited by two energized spark plugs. A pneumatically operated poppet valve, mounted on the main oxidizer valve (MOV), controls oxidizer flow to the ASI chamber.

The fuel turbopump is an axial-flow pump with direct turbine drive and consists of an inducer, a seven-stage rotor, and a stator assembly. It is a self-lubricated, high-speed pump with a two-stage turbine and supplies fuel to the thrust chamber at high pressure. A pressurized start tank supplies initial start energy to the turbine; a gas generator supplies a sustained energy source during engine operation. The fuel and oxidizer turbines are gas coupled by an 8-in.-diam crossover duct that directs gases exiting the fuel turbine to the oxidizer turbine inlet. The LH_2 enters the turbopump through an 8.0-in.-diam low pressure duct and exits through a 4.0-in.-diam high pressure duct. Nominal steady-state operating speed of the turbopump is 26,500 rpm at an engine mixture ratio of 5.5.

The oxidizer turbopump is a single-stage centrifugal pump with direct turbine drive. It is a self-lubricated, high-speed pump with a two-stage turbine and supplies LO_2 to the thrust chamber at high pressures. Gases discharged from the fuel turbine pass through the oxidizer turbine and exit through eyelets (total area of 115 in.^{2}) into the thrust chamber at an area ratio (A/A_t) ranging from 10.45 to 11.40. A percentage of fuel turbine exhaust gas volume is routed directly to the thrust chamber and bypasses the oxidizer turbine during the ES transient to prevent an overspeed of the turbine. When MOV opens to its first stage, the bypass valve is sequenced closed, and a greater volume of fuel turbine exhaust gases is directed to the oxidizer turbine. With the bypass valve closed, a nozzle in the valve bypasses a small percentage of gas and acts as a calibration device for turbopump performance balance and engine O/F ratio. The LO_2 is supplied to the pump through an 8.0-in.-diam low pressure duct and discharged through a 4.0-in.-diam high pressure duct. Nominal steady-state operating speed of the turbopump is 8500 rpm at an engine mixture ratio of 5.5.

The gas generator consists of a combustion chamber containing two spark plugs, a pneumatically operated control valve containing oxidizer and fuel poppets, and an injector assembly. The oxidizer and fuel poppets are mechanically linked by an actuator and provide a fuel lead to the gas generator combustion chamber. Spark exciters in the electrical control package provide energy to the spark plugs to ignite the propellants flowing through the open poppets into the combustion chamber.

The duct from the gas generator outlet directs the hot gases to the fuel and oxidizer turbines.

An engine-mounted, spherical tank is comprised of an integral GH_2 start tank and a He tank. The integral tank consists of a 7258-in.^{3} sphere for H_2 with a 1000-in.^{3} sphere for He located within it. Pressurized GH_2 in the fuel start tank provides the energy source for spinning the propellant turbines during the starting of the engine. The GH_2 for repressurization of the start tank during engine main-stage operation is obtained from the thrust chamber fuel manifold inlet and the fuel injection manifold.

The He tank provides a He pressure supply to the engine control system. A pneumatic accumulator, filled from the He tank, provides the necessary He for an emergency shutdown in case the He tank pressure is lost during engine operation.

The main fuel valve is a butterfly-type valve which is spring loaded in the closed position. The valve is pneumatically operated to the open position and pneumatically assisted to the closed position. It is mounted in the fuel high pressure duct between the fuel turbopump and the thrust chamber fuel manifold. A sequence valve mounted on the main fuel valve controls pneumatic pressure to the start tank discharge valve (STDV) solenoid control valve.

The MOV is a butterfly-type valve, spring loaded in the closed position. The valve, which opens in two stages, is pneumatically operated to the open position and pneumatically assisted to the closed position. It is mounted in the oxidizer high pressure duct between the oxidizer turbopump and the oxidizer injector manifold. A sequence valve on MOV controls pneumatic pressure to the gas generator control valve.

The electrically operated PU valve is a motor-driven valve that bypasses a percentage of LO_2 from the discharge side of the oxidizer turbopump to the pump inlet side. Nominal O/F ratio range control with the PU valve is 4.5 (full open) to 5.5 (full closed) over an angle position change of 57 deg. The purpose of the PU valve during flight application is to ensure the simultaneous exhaustion of the contents of the vehicle propellant tank. During engine flight, propellant level sensors in the vehicle propellant tanks control the valve position for adjusting oxidizer flow. At AEDC, the PU valve position was controlled from the J-4 Control Room.

An engine-mounted pneumatic control package controls pneumatics required for all pneumatically operated valves and engine-supplied purges. Valves controlled with the pneumatic package include the main fuel valve, first and second stages of MOV, the oxidizer turbine bypass valve, the ASI oxidizer valve, purge control valve, and the fuel and oxidizer bleed valves. Purges of He to the main injector oxidizer dome and the gas generator oxidizer injector are supplied from the pneumatic package.

An engine-mounted, electrical control package provides the logic required for proper sequence of the engine components during operation. The electrical control package contains a sequence controller, to properly sequence engine start and shutdown, and a spark ignition system that energizes the gas generator and ASI spark plugs.

Primary and auxiliary flight instrumentation packages contain sensors required to monitor critical engine parameters. The packages provide environmental control for the sensors.

2.1.2 S-IVB Battleship Stage

The S-IVB battleship stage (Fig. 9) has an internal volume of 10,456 and 2822 ft^3, respectively, for fuel and oxidizer, which provides an approximate propellant maximum capacity of 46,000 lb of LH_2 and 199,000 lb of LO_2. The stage has an internal diameter of 21.6 ft and is approximately 49 ft long. Propellant prevalves, in the low pressure ducts interfacing the stage and the engine, contain propellants in the stage until being admitted into the engine (to the main propellant valves) to chill the engine propellant supply systems. Propellant recirculation pumps in the vehicle tankage are utilized to circulate propellants through the low pressure ducts and turbopumps for hardware chilldown purposes before ES.

2.2 TEST CELL

Test Cell J-4 (Fig. 10) is a vertically oriented test unit designed for static testing of large liquid-propellant rocket engines and entire propulsion systems at pressure altitudes of 100,000 ft. The basic cell construction provides a 1.5 million-lb_f-thrust capacity. A general description of the features and operating principles of Test Cell J-4 in the subsequent paragraphs can more readily be presented using Fig. 10 as an orientation guide. Reference 1 provides a more detailed description of the test cell.

The installation of the J-2 engine/S-IVB battleship stage inside the test capsule orients the engine vertically downward on the centerline of the diffuser-ejector assembly. The basic dimensions of the test capsule are 48 ft in diameter and 81 ft in height. The diffuser insert (13.5 ft in diameter by 30 ft long) was sized for the J-2 engine to provide engine/diffuser pumping for meeting the program altitude requirements. The diffuser insert is mounted at the inlet of the 20-ft-diam by 150-ft-long diffuser. The centerbody steam ejector in the 20-ft-diam diffuser is provided for pre-fire pumpdown of the test capsule to the desired pressure altitude and for significantly reducing blowback at engine shutdown. The GN_2 annular ejector, mounted at the inlet of the diffuser insert, is provided to essentially eliminate steam blowback at steam ejector shutdown. A test capsule GN_2 repressurization system is provided to prevent steam blowback during the steam ejector operation transition and to repressurize the test capsule after engine shutdown.

A concrete spray chamber is provided directly beneath the test capsule to cool and dehumidify the rocket exhaust gas and steam ejector flow. This chamber is 100 ft in diameter and 250 ft deep. Suspended within the spray chamber are the diffuser-ejector system, a film-cooled exhaust deflector plate, three levels of water spray rings, and an LN_2 cell inerting and R-factor system. The lower 71 ft of the chamber (that portion below the bottom of the flame deflector) is used for spray water accumulation during a firing. During engine operation, the exhaust gases are collected by the diffuser for dynamic pressure recovery and discharged into the spray chamber. The deflector plate assists in turning the exhaust gas flow from the diffuser upward within the two million-ft^3 spray chamber. The expansion of the exhaust gases in this large volume results in further dynamic pressure recovery and therefore a reduction in their velocity. This low velocity, in combination with the long flow path to the three evacuation ports just beneath the spray chamber cap, causes a significant exhaust gas dwell time in the spray chamber. The water spray system (262,000-gpm) utilizes this dwell time to effectively cool and dehumidify the exhaust gases. This reduces exhaust mass flow rate by 85 percent. However, N_2 must be introduced at about 10 times the mass flow rate of H_2 in the exhaust gas for safety considerations and for reducing the gas R-factor value to a level acceptable to the rotating exhaust machinery. The three spray chamber evacuation ports route the cooled gases to a common duct which is connected directly to the exhaust machinery. This machinery provides compression of the gases to atmospheric pressure.

The test cell and exhaust ducting are inerted before the transfer of GH_2 or LH_2 to the stage and are maintained inert at all times while H_2 is aboard the stage. The test cell is considered inert provided the O_2 content is less than 4.9 percent by volume. An LN_2 system, injecting into the spray chamber, is utilized for inerting spray chamber and exhaust ducting. The water spray system is operated during the inerting process to ensure complete vaporization of N_2 after injection. The test cell is maintained inert by a GN_2 purge injected at the top of the test capsule. The purge flow rate, established at 3.2 times the air in-leakage rate, maintains the test cell O_2 content below the O_2-H_2 flammability limit.

Certain facility modifications were necessary to conduct this test program, the most significant being the addition of a testing capability for LH_2-fueled engines. The modifications involving the bulk of the design, fabrication (or construction), and installation time included (1) vehicle support stand system, (2) diffuser insert, (3) test capsule extension, (4) blast wall extension, (5) GN_2 annular ejector, (6) test cell inerting and R-factor system, (7) consumable storage facilities expansion, (8) test capsule inerting purge and repressurization systems, (9) engine environmental system, (10) electrical systems expansion, and (11) site preparation.

The engine/vehicle support stand consisted of a support frame, frame-to-stage adapter, and a side-force measuring system. The force measuring system was designed to measure engine pitch and yaw forces up to 20,000 lb_f (tension or compression) and for in-place calibration of the load cells.

A diffuser insert to the inlet of the existing 20-ft-diam diffuser was provided for dynamic pressure recovery of the engine exhaust gases to meet program altitude requirements. The basic dimensions of the water-cooled insert were 13.5 ft in diameter and 30 ft in length. A GN_2 annular ejector, designed for a maximum flow rate of about 500 lb_m/sec, was provided at the entrance to the diffuser insert to prevent steam blowback at steam ejector shutdown.

The consumable storage facilities were expanded to include LH_2, GH_2, GHe, and additional LN_2, GN_2, and steam (Fig. 11). The steam capacity was extended by installing a tieline between Test Cells J-3 and J-4 accumulators. The existing oxidizer and water storage systems were adequate.

The extension of the test capsule height to 80 ft was necessary to enclose the engine/stage installation. This extension was accomplished by the addition of two 16-ft-high, 48-ft-diam spool sections. The reinforced concrete blast wall, surrounding approximately 280 deg of the capsule, was extended by 20 to 60 ft above grade.

The test cell inerting and R-factor system was comprised of a LN_2 injection system in the spray chamber to reduce the gas constant and provide an inert atmosphere for the fuel-rich engine exhaust gases. The GN_2 test capsule repressurization system was provided for raising the test capsule pressure at engine shutdown to minimize blowback from the spray chamber and for a test capsule emergency purge in the event of a major propellant leak. The test capsule inerting purge system was provided for inerting the air in-leakage with vaporized LN_2.

The expansion of the electrical systems was required to provide the necessary electrical controls and power to the test cell complex. These included modification to existing power distribution centers, considerable wiring for control, electrical power, communication, and test area lighting systems, and a cable trench from the control room to test area.

Site preparation consisted of an extension of roadway and railroad networks for access to consumable storage system installations, consumable transporters, maintenance, and fire fighting equipment.

2.3 INSTALLATION

The test article installation was comprised of a J-2 engine/S-IVB battleship stage installed vertically (engine oriented downward on the diffuser-ejector assembly) in the test capsule (Figs. 10 and 12). Modifications to the test cell complex (Section 2. 2) were necessary for the test article installation and operation.

The installation of the engine and stage was accomplished by removing the top sections of the capsule. The stage was installed on the adapter section of the vehicle support stand; the engine was attached to the stage at the gimbal point. The vehicle and engine propellant systems were interfaced by installing the low pressure propellant ducts, and facility propellant fill, dump, vent, and purge systems were mated with the stage fuel and oxidizer tanks. The pitch and yaw thrust measurement columns were connected to the thrust chamber outrigger assemblies, and the modular universal flexures were aligned to permit the forces to be transferred to the load cells with interactions less than ± 200 lb_f. Facility-supplied engine and stage purges, pneumatics, electrical control systems, and instrumentation were installed, and the engine-to-stage systems were interfaced.

2.4 INSTRUMENTATION

Instrumentation systems were provided to measure the engine, stage, and facility parameters of interest. Location and designation of selected engine- and stage-associated instrumentation used throughout this program are presented in Fig. 13 and Table III, respectively. The engine instrumentation was comprised of flight- and facility-supplemented instrumentation. The flight instrumentation, which measures critical engine parameters, was provided by the engine manufacturer and located in primary and auxiliary instrumentation packages on the engine (Fig. 5). The facility-supplemented engine instrumentation was provided to verify the flight instrumentation data and to measure additional engine parameters.

All pressure measurements were made using strain-gage-type pressure transducers. The facility-supplied transducers were laboratory calibrated at AEDC against a secondary standard before installation; the flight and many of the stage instrumentation transducer calibrations were provided by the engine and stage manufacturers, respectively. Electrical calibrations, consisting of substituting precision electrical shunt resistances, were used to calibrate the data acquisition systems. Periodic in-place calibrations of selected transducers used to monitor pressures specified as test requirements were accomplished.

Temperature measurements were made using resistance temperature transducers (RTT) and several types of thermocouples. The RTT units were used primarily for measuring the temperatures of the propellants and various engine components. A combination of copper-constantan, iron-constantan, and Chromel®-Alumel® thermocouples was used to measure engine hardware and facility temperatures. The data acquisition systems for the RTT and thermocouple measurements were calibrated using precision electrical resistance and voltage substitutions, respectively. Data reduction of RTT measurements was obtained from the manufacturer's calibration data. The thermocouples were referenced to a 150°F junction.

Engine side loads were measured using dual-bridge, strain-gage-type load cells. The load cells were laboratory calibrated before installation. The recording systems were calibrated with precision electrical shunt resistances before each test. The measurement of axial thrust was not a program requirement.

Oxidizer and fuel turbopump shaft speeds were sensed by a magnetic pickup. Each pump produces a frequency output of 12 pulses per shaft revolution. The calibration of the recording systems was accomplished by using frequency substitutions.

Engine fuel and oxidizer flow rates were measured with turbine-type flowmeters. The frequency output of the fuel and oxidizer flowmeters was four and six pulses per revolution, respectively. The fuel and oxidizer flow rates through the propellant recirculation system were also monitored with turbine-type flowmeters. Flowmeter calibrations were supplied by the engine and stage manufacturer. The associated recording systems were calibrated using frequency substitution.

Vibrations (vertical plane) produced during engine operation were measured by accelerometers mounted on the oxidizer injector dome. Accelerometers on the fuel and oxidizer turbopumps sensed vibrations in a horizontal plane. The accelerometers were laboratory calibrated at AEDC by shaker tests before installation. Calibrations of the associated recording systems consisted of frequency-voltage substitutions.

Selected engine valves were instrumented by the engine manufacturer with linear potentiometers and microswitches.

The types of data acquisition and recording systems used during this program were (1) a multiple-input digital data acquisition system (MicroSADIC®) scanning each parameter at 40 samples per second and recording on magnetic tape, (2) single-input, continuous-recording, analog system (Vidar®) recording on magnetic tape, (3) single-input, continuous-recording, FM systems recording on magnetic tape, (4) photographically recording galvanometer-type oscillographs, (5) direct inking, null-balance, potentiometer-type X-Y plotters and strip charts, and (6) optical data recording. These systems were calibrated as required before each test (atmospheric and altitude calibrations) for each parameter measured. Television cameras were used to provide visual coverage during an engine firing.

SECTION III
SEQUENCE

Facility logic interfaced with engine logic provided the necessary control to (1) start the engine, (2) sequence the engine properly during start and shutdown transients, and (3) terminate a firing in the event of an engine or vehicle stage malfunction.

3.1 FACILITY LOGIC

Facility logic was interfaced with engine logic to monitor and control the engine and stage start and shutdown sequence. The facility logic was controlled from an automatic countdown sequencer programmed by paper tape input. The sequencer was programmed to ascertain firing readiness by sampling system-ready switches 1 sec before fire command. Approximately 0.2 sec before sequencer "zero", the sequencer initiates a fire command signal which energizes the facility logic; at sequencer zero, the sequencer is programmed to stop until the STDV solenoid is energized.

The propellant prevalves are commanded open at fire command by the facility logic. After both prevalves are open, a 5-sec propellant recirculation "off" timer is energized. A sequence monitor timer is set to terminate the countdown if the combined time required from fire command to expiration of the 5-sec timer exceeds 13 sec. At expiration of the 5-sec timer, an ES command is initiated, a facility fuel lead timer is energized, and a facility "start OK" timer is energized. The ES command signal is terminated after 200 msec by a facility timer; at this time, the start command is locked in by the engine logic. The facility start OK timers are provided as engine backup timers to automatically terminate a firing if the engine "main-stage pressure OK" signal has not been obtained within a predetermined time period. If, after expiration of the fuel lead timer, the engine starting conditions are satisfied, an open signal will be applied to the STDV solenoid. This signal, which causes the sequencer to restart, is used as engine and sequencer zero reference.

At expiration of run duration, an engine shutdown command is initiated by the facility sequencer. The shutdown command activates the engine and facility cutoff logic. This starts facility-supplied cutoff purges and closes the propellant prevalves.

Selected critical engine parameters are automatically monitored during a firing with the engine safety cutoff system (ESCS) and manually monitored by observers in the control room. Emergency shutdown circuitry is monitored at all times by the facility logic system and initiates an engine shutdown command if any malfunctions occur.

The ESCS was provided to monitor critical engine parameters and to initiate a shutdown command, if predetermined values are exceeded. Parameters monitored were (1) fuel turbine overspeed, (2) engine vibrations, and (3) the gas generator outlet temperatures (GGOT).

An automatic shutdown limit of 28,400 rpm was established as the maximum allowable fuel turbine speed during engine main-stage operation. The fuel turbine overspeed requirement was deleted after test 03.

Sustained engine vibrations in excess of 150 g for greater than 70 msec were established as the automatic "kill" limit through test 05. After test 05, the timer was reset to 150 msec.

Automatic shutdown limits established for GGOT are presented in Fig. 14. The minimum allowable GGOT limit was 250°F at approximately 1.24 sec after the STDV solenoid energized. The maximum allowable limit was 2000°F from approximately 1.24 to 3.94 sec after the STDV solenoid energized. A maximum allowable temperature for steady-state operation was 1450°F.

3.2 ENGINE SEQUENCE

3.2.1 Engine Start Sequence

An engine schematic and an operating sequence diagram are presented in Figs. 15 and 16. The ES can be initiated after all engine, vehicle, and facility-ready systems have been made, and thereafter each event is sequenced by an engine sequence controller. Initiation of ES command (facility-initiated) activates the ES module, which simultaneously opens the He control valve, the ignition phase control valve, and energizes the spark plug exciters. The start tank discharge control timer and fuel lead timer are also energized at ES command. The He control valve regulates He flow to fill the pneumatic accumulator, to close the propellant bleed valves, and to purge the oxidizer dome and gas generator oxidizer injector manifold through the purge control valve. The ignition phase control valve routes He pressure to open the ASI oxidizer valve and the main fuel valve and supplies pressure to the inlet port of the sequence valve located within the MOV first-stage actuator. A check valve in the pneumatic control package ensures continued pressure to engine valves in the event of a He supply failure. Once the ASI oxidizer valve and the main fuel valve are opened, propellants flow under static head to the ASI chamber and are ignited. When the main fuel valve is 90 percent open, a sequence valve opens and supplies pneumatic opening pressure to the STDV solenoid control valve.

A normal engine sequence will continue with the opening of STDV and the energizing of the ignition phase timer (450-msec timer) if the following conditions exist (1) the main fuel valve and fuel sequence valve are open, (2) proper fuel quality at the injector is verified by a fuel injector temperature below -150°F, (3) start tank discharge control timer has expired (640-msec timer initiated at engine start), and (4) the fuel lead timer (facility timer) has expired. With these four conditions satisfied, STDV opens to release GH_2 to the fuel and oxidizer turbine. Once the ignition phase timer has expired, STDV is closed by de-energizing the control solenoid, a 3.3-sec spark plug de-energize timer is activated, and the main-stage control module is energized. If ignition has not been detected (as indicated by the ignition detect probe in ASI) when the ignition phase timer has expired, engine cutoff will occur. After the main-stage control module is energized, the main-stage control valve opens, venting He pressure from the MOV closing actuator and the opening port of the purge control valve. The purge control valve closes, and the oxidizer dome and gas generator oxidizer purges are terminated. Opening pressure is applied to the MOV opening actuators, and the MOV first stage opens. A sequence valve in MOV opens and supplies pressure to open the gas generator control valve and close the oxidizer turbine bypass valve. Fuel and oxidizer flow to the gas generator are controlled by poppets in the gas generator control valve that open sequentially to provide a fuel lead. Gases generated are directed in series to the fuel and then oxidizer turbines. The second stage of MOV starts opening approximately 0.6 sec after the main-stage control valve is opened. The second-stage valve ramp time is controlled by venting closing pressure through an orificed check valve. As the propellant turbopumps approach steady-state operation, the main-stage pressure OK signal is generated (by a pressure switch), and steady-state engine operation follows. If the main-stage pressure OK signal has not been initiated before the sparks de-energize timer expires, engine cutoff will occur. The time from ES command to main-stage pressure OK signal is primarily dependent on the fuel lead time.

3.2.2 Engine Shutdown Sequence

A cutoff signal applied to the sequence controller simultaneously de-energizes the control solenoids for closing the main-stage control and ignition phase control valves and energizes the He control de-energize timer. Opening control pressure for the MOV actuator, ASI oxidizer valve, and main fuel valve is vented through ports in the pneumatic control package. The He pressure is supplied to the closing actuator of MOV, the opening control port of the purge control valve, closing actuator of the ASI oxidizer valve, closing actuator of the main fuel valve, and the opening control port of the pressure-actuated fast shutdown control valve. Oxidizer dome and gas generator oxidizer line purges begin as soon as thrust chamber and gas generator pressures drop below the He control pressure level. With the exception of the ASI oxidizer valve, propellant bleed valves, and the oxidizer turbine bypass valve, all valves are spring loaded to the closed position. The oxidizer turbine bypass valve is spring loaded to the open position and starts to open as soon as closing pressure is vented. When the He control de-energize timer expires, the He control valve closes, and control system pressure is vented through the oxidizer dome and gas generator oxidizer line purges. When He control pressure drops to the actuation pressure level of the normally closed purge control valve, the valve closes, and the purges stop. Closing pressure to the propellant bleed valves is bled off, and these valves open under spring pressure.

SECTION IV
PROCEDURE

The major facility modifications necessary to conduct this program required many subsystem checkouts. As subsystems were brought to an operable state, these were integrated with the complete system and checked out to assure proper operation. After the S-IVB stage and J-2 engine were installed in the test cell, all required electrical and mechanical hookups were accomplished. Final facility, engine, and stage readiness for a firing was verified in an integrated systems checkout. This checkout, identified as test number J4-1554-01, was conducted on July 30, 1966. All functions required to fire the J-2 engine were accomplished. A firing was not planned for this test period.

Pre-operational procedures were begun several hours before each test. All consumable storage systems were replenished, and engine inspections and leak checks were conducted. Chemical analysis of propellants, propellant tank pressurants, and engine pneumatic and purge gas samples was made to ensure that test specifications were met. Facility and engine sequence checks and engine abort checks were conducted within a 24-hr time period before an engine firing to verify the proper sequence of events. The abort checks consisted of electrically simulating engine malfunctions to verify the occurrence of a cutoff signal in the event of such a malfunction during engine operation. Engine and facility sequence checks consisted of verifying the timing of all engine and facility valves and events to be within specified limits. A final engine sequence check was conducted immediately preceding the cell evacuation for a test period.

Upon completion of atmospheric instrumentation calibrations, the test cell was evacuated to approximately 0.5 psia with the exhaust machinery; altitude instrumentation calibrations were conducted; and subsequently, a cell air in-leakage evaluation was performed. Oxidizer dome, gas generator oxidizer injector, and thrust chamber jacket purges were initiated before evacuating the test cell (engine purges required for a typical test period are presented in Table IV). Immediately before loading propellants on board the vehicle, the cell and exhaust ducting atmosphere was inerted with approximately 20,000 lb of N_2 to reduce the O_2 content to less than 4.9 percent by volume (lower flammability limit of H_2-O_2 mixtures). At this same time, the cell N_2 purge was initiated at a rate equivalent to the cell air in-leakage multiplied by 3.2. This cell purge (6- to 10-lb/sec) continuously inerted the air leaking into the cell for the duration of the test period. The vehicle propellant tanks were then loaded to the 30-percent level (a test requirement maximum), and the remainder of the terminal countdown was conducted. A typical countdown was as follows:

Time	Event
t_o - 6 hr	1. Initiate oxidizer dome and gas generator oxidizer injector He purges. 2. Initiate low level thrust chamber jacket He purge. 3. Evacuate test cell.
t_o - 5 hr, 45 min	1. Cut off oxidizer dome and gas generator oxidizer injector He purges. 2. Initiate oxidizer dome GN_2 purge.
t_o - 5 hr, 30 min	Calibrate instrumentation at altitude conditions.
t_o - 4 hr, 45 min	Evaluate test cell air in-leakage.
t_o - 4.00 hr	1. Increase thrust chamber jacket He purge to high level. 2. Inert cell with N_2. 3. Initiate cell GN_2 purge.
t_o - 3 hr, 45 min	1. Reduce thrust chamber jacket He purge to low level. 2. Transfer propellants to the vehicle tanks.
t_o - 1 hr, 10 min	Initiate a 10-min turbopump and gas generator He purge.
t_o - 1 hr	1. Increase thrust chamber jacket He purge to high level. 2. Open fuel and oxidizer prevalves and temperature condition propellants.

Time	Event
t_o - 51 min	Initiate start tank purge.
t_o - 50 min	1. Cycle fuel and oxidizer tank vent valves.
	2. Open propellant recirculation valves.
t_o - 46 min	Condition the fuel start tank.
t_o - 45 min	Reduce thrust chamber jacket He purge to low level.
t_o - 15 min	1. Cut off thrust chamber jacket low level He purge.
	2. Temperature condition the thrust chamber.
	3. Cut off oxidizer dome GN_2 purge.
t_o - 660 sec	Initiate remainder of countdown on automatic sequence timing.
t_o - 600 sec	Start propellant recirculation pumps.
t_o - 590 sec	Close fuel and oxidizer prevalves.
t_o - 240 sec	Initiate turbopump and gas generator He purges.
t_o - 200 sec	Initiate partial spray chamber water coolant flow.
t_o - 150 sec	Pressurize the oxidizer vehicle tank.
t_o - 130 sec	1. Increase cell GN_2 purge to maximum flow.
	2. Begin cell GN_2 repressurization flow maximum rate.
t_o - 120 sec	1. Initiate LN_2 flow at 50 lb/sec.
	2. Pressurize the fuel vehicle tank.
t_o - 100 sec	Cut off turbopump and gas generator He purges.
t_o - 80 sec	1. Increase MicroSADIC to maximum sampling rate.
	2. Start analog strip recorders.
	3. Turn on cell photograph lights.
	4. Initiate remainder of spray chamber water coolant flow.
t_o - 70 sec	Start the motion-picture cameras.
t_o - 60 sec	Initiate a 20-sec ramp of steam ejector driving pressure to initial level (100-psia).

Time	Event
t_o - 25 sec	1. Cut off cell GN_2 purge.
	2. Cut off cell GN_2 repressurization flow.
	3. Increase LN_2 flow to 75 lb_m/sec
t_o - 20.5 sec	Automatic hold, if exhauster plant and coolant flow systems are not ready to proceed into firing.
t_o - 10 sec	Initiate high-speed analog recorders.
t_o - 1 sec	Automatic hold if control systems and/or start requirements are judged unsatisfactory for firing.
t_o	1. Initiate fire command. The sequencer is programmed to hold until the engine STDV solenoid is energized and then resumes the count.
	2. Increase LN_2 flow to 300 lb_m/sec
t_o + 59 sec	Initiate GN_2 ejector flow of 475 lb_m/sec.
t_o + 60 sec	1. Initiate engine cutoff signal.
	2. Initiate maximum cell GN_2 interdiffuser and repressurization flow.
t_o + 62 sec	Cut off deflector plate coolant water.
t_o + 65 sec	Initiate 25-sec cutoff ramp of steam ejector driving pressure.
t_o + 75 sec	Cut off motion-picture cameras.
t_o + 80 sec	1. Initiate 15-sec cutoff ramp of GN_2 ejector flow.
	2. Cut off cell photograph lights.
t_o + 85 sec	1. Reduce MicroSADIC sampling rate.
	2. Cut off high-speed analog recorders.
t_o + 95 sec	1. Reduce spray chamber water coolant flow.
	2. Reset engine firing panel - de-energize engine ignition and control power buses.
t_o + 100 sec	Vent vehicle fuel and oxidizer tanks.
t_o + 200 sec	Cut off LN_2 flow,
t_o + 210 sec	Cut off remaining spray chamber water coolant.
t_o + 300 sec	Initiate vehicle fuel and oxidizer detanking.
t_o + 370 sec	Cut off GN_2 interdiffuser purge.

Time	Event
t_o + 660	1. Cut off turbopump and gas generator He purges.
	2. Cut off oxidizer dome GN_2 purge.
	3. Cut off thrust chamber jacket He purge
t_o + 15 min	Vent and purge fuel start tank.
t_o + 1 hr	Initiate purging and inerting of vehicle fuel and oxidizer tanks after detanking is completed.
t_o + 2 hr	Initiate post-fire cell N_2 inerting.
t_o + 2 hr, 15 min	Conduct post-fire altitude instrumentation calibrations.
t_o + 3.00 hr	Return test cell pressure to atmospheric level.

SECTION V
RESULTS AND DISCUSSION

5.1 GENERAL

The objectives of the first 11 test periods of the J-2 Engine EVT Program are outlined below:

1. Evaluate the engine transient operation and performance at a pressure altitude of approximately 100, 000 ft.
2. Acquire data to better define the engine operating conditions and limits for flight applications.

These objectives were accomplished primarily utilizing the start conditions for the J-2 engine Saturn IB flight applications. However, some tests included objectives in support of the J-2 engine restart application – the S-IVB stage of the Saturn V.

Test conditions and results and selected engine and stage performance parameters are summarized in Tables V through VII. The test matrix outlining the required target conditions for ES is presented in Table VIII. All ES were accomplished with the PU valve in null position (O/F ratio of 5. 0).

Since the data presentation will be directed toward detailed discussion of engine systems operation, pertinent comments for each test period will be presented initially. Optimum facility steam ejector performance was not realized on earlier tests. This was because proper sequencing of facility systems was being developed during these tests and resulted in the engine being subjected to a low pressure altitude and a steam environment at ES. Three firings (05, 06, and 09A) were conducted without facility steam ejector operation. Firing durations were progressively increased, consistent with the development of facility operational procedures, to a maximum of 40 sec in working toward a 60-sec firing goal (a 60-sec firing duration is required for the engine to attain thermal equilibrium).

5.1.1 J4-1554-01

Test Date	July 30, 1966

Test 01 was a readiness and reliability checkout of all engine, stage, and facility systems and involved an integrated countdown of all systems. An engine firing was not scheduled; however, all functions necessary for a normal firing were accomplished.

The engine MOV assembly was replaced before test 02 to incorporate a temperature compensating orifice.

5.1.2 J4-1554-02

Test Date	August 26, 1966
Pressure Altitude at ES, ft	67,000 (Ref. 3)
Firing Duration, sec	1.169

The programmed 20-sec engine firing was terminated prematurely by ESCS, which sensed a fuel turbine overspeed. Analysis of the fuel turbine speed data revealed the occurrence of a faulty engine cutoff since turbine speed was well within the spinup specifications. The source generating the cutoff signal could not be isolated. However, provisions were made to record the output signal from the fuel turbine tachometer on subsequent tests to investigate the possibility of its generating the signal level for cutoff.

A maximum GGOT of 1700°F occurred. This peak temperature was considerably higher than observed for engine acceptance tests at lower engine ambient altitude conditions. Sporadic vibration safety counts (VSC) were observed for 30 msec; no VSC were expected. Maximum chamber pressure attained was 80 psia.

5.1.3 J4-1554-03

Test Date	September 2, 1966
Pressure Altitude at ES, ft	77,400
Firing Duration, sec	1.052

The ESCS again sensed a fuel turbine overspeed and prematurely terminated the engine firing scheduled for 20-sec duration. The fuel turbine speed was again within the spinup specifications, which indicated the reoccurrence of a faulty engine cutoff. The analysis of wave form of the turbine tachometer output signal revealed two peaks per cycle, both of which were being counted by the ESCS overspeed trip circuitry. The double counting of the true turbine output signal frequency was concluded as the source of the faulty engine cutoff for this test and for test 02. As a result, this automatic shutdown circuit was deleted from ESCS for subsequent tests.

An extremely high GGOT spike of 2230°F occurred before engine shutdown. Engine VSC were encountered for 12 msec. Maximum chamber pressure attained was 137 psia.

5.1.4 J4-1554-04

Test Date	September 14, 1966
Pressure Altitude at ES, ft	67,000
Firing Duration, sec	1.082

The programmed 20-sec firing was terminated by ESCS for excessive VSC duration. Engine cutoff occurred after 112 msec of sporadic VSC; however, the VSC duration continued for 20 msec during engine shutdown. A maximum GGOT of 1880°F was recorded at engine shutdown. Maximum chamber pressure attained was 145 psia.

Since there were possibilities of injector icing resulting from the engine environment at ES, an improved oxidizer dome purge diffuser was installed, and the purge diffuser GN_2 supply was heated.

5.1.5 J4-1554-05

Test Date	September 24, 1966
Pressure Altitude at ES, ft	53,000
Firing Duration, sec	10.075

A 10-sec firing was successfully conducted to achieve the first main-stage operation of the J-2 engine at AEDC. Engine 90-percent performance level (thrust chamber pressure of 600 psia) was attained at 2.108 sec. Sporadic VSC were experienced for 67 msec. A peak GGOT of 1860°F was recorded. Side forces produced during the engine transient operation were considerably reduced from engine acceptance tests and were within acceptable limits. The automatic shutdown limit for sustained VSC was changed from 70 to 150 msec for subsequent tests.

5.1.6 J4-1554-06

Test Date	September 30, 1966
Pressure Altitude at ES, ft	58,000
Firing Duration, sec	10.075

The engine firing was successfully conducted for the programmed 10 sec. Engine 90-percent performance level was attained at 2.133 sec. Engine VSC were not observed. A peak GGOT of 1740°F was recorded.

5.1.7 J4-1554-07

Test Date	October 7, 1966
Pressure Altitude at ES, ft	72,800
Firing Duration, sec	20.070

The programmed 20-sec firing was successfully accomplished. Engine 90-percent performance level was attained at 2.133 sec. Sporadic VSC were recorded for 30 msec; a peak GGOT of 1650°F was indicated. A malfunction in the prechill controller system permitted the ignition phase timer to be enabled, even though the fuel injector temperature (-74°F) was considerably warmer than the level (-150°F) required. A faulty fuel injector temperature probe produced this anomaly. It is significant that no fuel pump low-speed stall occurred because of the rapid thrust chamber chilldown during the ES transient at altitude conditions.

The oxidizer injector dome purge diffuser was replaced with a standard ground support equipment (GSE) diffuser before test 08.

5.1.8 J4-1554-08

Test Date	October 18, 1966
Pressure Altitude at ES, ft	75,000
Firing Duration, sec	20.078

The engine firing was successfully conducted for the programmed 20 sec. Engine 90-percent performance level was attained at 2.225 sec. Sporadic VSC were recorded for 15 msec. Peak GGOT of 1740°F was indicated.

5.1.9 J4-1554-09

Test Date	October 27, 1966	
	Test 09A	Test 09B
Pressure Altitude at ES, ft	62,000	93,000
Firing Duration, sec	5.068	0.454

The programmed 5-sec engine firing for test 09A was successfully accomplished. Engine 90-percent performance level was attained at 2.108 sec. Engine VSC were recorded for 3 msec. The peak GGOT was 1700°F. Before test 09A, an engine chilldown procedure was evaluated which involved temperature conditioning the hardware associated with the propellant low pressure ducts and turbopumps using the vehicle propellant recirculation systems.

Test 09B, programmed for 0.4-sec duration and including an 8-sec fuel lead with no prior thrust chamber temperature conditioning, was successfully conducted in support of an engine restart study. The LH_2 temperature at the fuel injector after the 8-sec fuel lead was the deciding factor for a 4.5-sec fuel lead on test 10B.

5.1.10 J4-1554-10

Test Date	November 4, 1966		
	Test 10A	Test 10B	Test 10C
Pressure Altitude at ES, ft	102,500	109,000	109,000
Firing Duration, sec	5.070	30.068	N/A

The programmed 5-sec engine firing for test 10A was successfully accomplished. Engine 90-percent performance level was attained at 2.100 sec. Engine VSC were recorded for 23 msec. The transient GGOT indicated two peaks of significance. The initial peak was 1910°F; the second peak was 1880°F.

The engine firing for test 10B was successfully accomplished for the programmed 30-sec duration. Engine 90-percent performance level was attained at 2.116 sec. This test was the initial engine restart test. Engine VSC were not observed. The GGOT again exhibited two transient peaks. The initial peak was 1590°F; the second peak was 1680°F. The second GGOT peak was the first GGOT overshoot experienced.

Test 10C, which did not involve an engine firing, was conducted to evaluate an engine/stage chilldown procedure for temperature conditioning the hardware associated with the propellant low pressure ducts and turbopumps for engine restart.

5.1.11 J4-1554-11

Test Date	November 16, 1966		
	Test 11A	Test 11B	Test 11C
Pressure Altitude at ES, ft	108,000	105,500	108,700
Firing Duration, sec	40.070	1.378	0.071

The programmed 40-sec engine firing for test 11A was successfully accomplished. Engine 90-percent performance level was attained at 2.100 sec. This test was the first engine firing during this program with a PU valve excursion. The PU valve excursion, at approximately 5 sec, effectively changed the engine mixture ratio from 5.0 to 5.5. Sporadic VSC were indicated for 83 msec. Two transient GGOT peaks were observed; the initial peak was 1790°F, and the second peak was 1730°F.

Test 11B, programmed for a 5-sec engine firing duration, was terminated prematurely by ESCS for a GGOT transient overtemperature. A GGOT overshoot temperature of 2150°F exceeded the automatic cutoff limit; the initial GGOT peak was 2000°F. Engine VSC were recorded for 20 msec. This was the first engine firing attempt with an 8-sec fuel lead.

Test 11C, which did not involve an engine firing, was conducted to evaluate an engine/stage chilldown procedure for temperature conditioning the hardware associated with propellant low ducts and turbopumps for engine restart.

5.2 ENGINE TRANSIENT OPERATIONS

The J-2 engine transient operations during this program revealed characteristics somewhat different from the engine acceptance tests and provided an important insight into the engine transient operation for the Saturn IB and Saturn V flights. The presentation will first be devoted to establishing relationships between the basic parameters for start transient analysis by using data from test 10B and then followed by discussions of the transient gas generator operation, thrust chamber side forces, combustion instability, fuel pump transient performance (head-flow characteristics), fuel lead effects, and engine shutdown transient.

5.2.1 Typical ES Transient

The relationship between basic engine parameters is established in this start transient section utilizing test 10B data. Data reference time is STDV solenoid energized. Before the reference time, the ES command had been initiated and the main fuel valve opened. Start transient traces for valve operations and basic engine parameters are presented in Fig. 17. Selected engine valve operating times for all firings are shown in Table IX.

The spinup characteristics of the fuel and oxidizer turbines during ES are presented in Fig. 18a. As initial opening of STDV occurs, start tank pressure supplies energy to both turbines. Initial STDV movement for test 10B occurred at 0.158 sec and was fully open at 0.315 sec. A peak oxidizer turbine speed of approximately 3550 rpm was observed to occur at 0.7 sec as a result of start tank blowdown. Fuel turbine speed continually increased, although at a reduced rate after 0.6 sec. The STDV started closing at 0.535 sec and was fully closed at 0.840 sec. An increase in start tank pressure was noted at 1 sec after STDV as a result of normal repressurization. Initial gas generator combustion occurred at 0.621 sec and continued the supply of operating energy for the turbines. Typical start transient fuel and oxidizer turbine inlet temperatures and the oxidizer turbine inlet pressure are shown in Fig. 18b. The fuel pump turbine inlet pressure was not measured but was similar to that of the gas generator chamber pressure, also shown in Fig. 18b.

The initial gas generator chamber pressure and fuel injector pressure spikes (Fig. 18c) reflect start tank blowdown. The oxidizer injector pressure initially reflects a He purge, which is terminated 0.44 sec after the STDV solenoid is energized. A check valve in the oxidizer injector prevents backflow when chamber pressure is higher than the injector pressure. Fuel and oxidizer flow to the gas generator is supplied through poppets that are sequentially opened. Initial opening of the fuel poppet occurred at 0.522 sec and was fully open at 0.575 sec; the oxidizer poppet started to open at 0.621 sec and was fully open at 0.697 sec. As can be observed in Fig. 18c, gas generator chamber pressure sharply increased when the oxidizer poppet was opened, indicating initial combustion. Note that at approximately 0.85 sec, gas generator chamber pressure exhibited an increase, whereas the oxidizer injector pressure decreased. The gas generator chamber pressure rise resulted from the gradual increase of fuel and oxidizer pump discharge pressures. Apparently the drop in injector pressure resulted from chilldown of the oxidizer bootstrap line (supply line from oxidizer pump discharge to the gas generator injector) resulting in improved oxidizer quality supplied to the gas generator. When thrust chamber ignition occurred (arbitrarily defined in this report as the time that thrust chamber pressure attains 100 psia), a higher resistance to the main fuel and oxidizer flows was developed. Consequently, a rapid increase in fuel pump discharge and gas generator fuel injector pressures occurred (Figs. 18c and d). However, only a gradual increase in oxidizer pump discharge and gas generator oxidizer injector pressures occurred (Figs. 18c and d) because of resistance being afforded to the oxidizer flow by MOV. As a result, the increase in fuel flow to the gas generator exceeded the increase in oxidizer flow and thereby reduced the gas generator O/F ratio and GGOT. Gas generator temperatures recorded for test 10B are shown in Fig. 18c. The peak initial temperature was 1590°F, and the overshoot was 1680°F.

The buildup in fuel pump discharge pressure during start tank blowdown (Fig. 18d) reflected turbine speed buildup. After opening the fuel poppet, the sudden pressure rise noted in fuel pump discharge pressure at approximately 0.55 sec apparently resulted from fuel backflow through the gas generator fuel poppet. An enlarged view, from 0.5 to 0.8 sec, of fuel pump discharge pressure and gas generator chamber and injector pressures is shown in Fig. 18e. Ten firings appeared to have fuel backflow. Another pressure spike in fuel pump discharge pressure at approximately 0.64 sec (Fig. 18d) appears to be backflow of hot combustion gases through the fuel poppet. The apparent backflow of hot gases was unique to test 10B. With the exception of test 10B, pump discharge and injector pressures were high enough to prevent hot gas backflow through the fuel poppet. Thrust chamber ignition, which occurred at 0.973 sec, increased the backpressure by developing resistance to propellant flow and caused a rapid rise in the fuel pump discharge pressure.

Oxidizer pump discharge pressure exhibited a rapid increase (Fig. 18d) with turbine spinup. The sharp increase resulted from high oxidizer system resistance caused by the MOV position. The first stage of MOV started opening at 0.494 sec and was fully open at 0.549 sec. From 0.60 to 0.85 sec, little change was noted in the oxidizer pump discharge pressure. At 0.85 sec the oxidizer turbine bypass valve was approximately 84 percent closed and caused a higher percentage of gas generator exhaust gases to be directed to the oxidizer turbine. The rate of increase of the oxidizer pump discharge pressure exceeds that of the fuel pump discharge pressure from approximately 1.00 to 1.20 sec. With the opening of the MOV second stage, the fuel pump discharge pressure began to converge on the oxidizer pump discharge pressure (Fig. 18d) until about 1.37 sec; afterwards, the rise rates of both pressures remained relatively constant until 1.80 sec. At this time, the pump discharge pressures were approaching steady-state levels.

The main fuel injector and thrust chamber pressure exhibited a low rate of increase until thrust chamber ignition occurred (Fig. 18f), at which time pressures increased to approximately 180 and 160 psia, respectively. Both pressures attained steady-state level at about 2.5 sec. Responses of oxidizer injector pressure measurements were inadequate for transient operation study and are not presented.

Fuel and oxidizer flow rates to the engine are presented in Fig. 18f. Fuel flow rate attained approximately 61 percent of steady state by 0.6 sec after the STDV solenoid was energized. Because of the oxidizer valve operating sequence, increase in oxidizer flow was relatively slow.

5.2.2 Gas Generator Observations

Start transient GGOT recorded at AEDC were consistently higher than observed during acceptance tests. This was attributed primarily to an improved oxidizer quality supply to the gas generator and lower fuel system resistance at altitude conditions during ES. The GGOT experienced have exhibited two peaks; an initial peak occurring approximately 1 sec after the STDV solenoid was energized and a second peak approximately 0.3 sec later. Since GGOT is a function of propellant O/F ratio in the gas generator, any factor changing fuel or oxidizer flow rate to the gas generator will affect GGOT. Items investigated as possible contributors to the magnitude of the initial peak were (1) fuel system flow resistance, which varies primarily with thrust chamber temperature conditions, (2) the fuel start tank energy, and (3) turbine hardware temperatures (including the crossover duct temperature). These factors, as well as the effect of the MOV opening sequence, were investigated as possible contributors to the GGOT second peak.

5.2.2.1 Fuel System Resistance Effect on GGOT, Initial Peak

A comparison of tests 05 and 07 demonstrates the effect of fuel system resistance on the initial GGOT peak. Fuel system resistance to flow increases as the thrust chamber temperature is increased. Using this criteria, fuel flow resistance was high for test 07 and low for test 05. Specific throat and exit temperatures at ES were:

Test No.	Throat Temperatures, °F	Exit Temperatures, °F	Resistance to Fuel Flow
5	-233	-216	Low
7	-88	-70	High

Start tank energy and oxidizer turbine hardware temperature compared closely for these two tests and hence are considered not to bias the initial GGOT peak.

Since the gas generator O/F ratio is dependent on fuel and oxidizer injector pressures, factors affecting these pressures were investigated. The initial start transient fuel pump discharge and gas generator fuel injector pressures (Figs. 19a and b) were observed to increase significantly with the warmer thrust chamber temperatures for test 07. The oxidizer pump turbine speed (Fig. 19c), pump discharge pressure (Fig. 19a), and gas generator injector pressure (Fig. 19b) were not significantly affected by the difference in fuel system resistance until about 0.7 sec after STDV. At approximately 0.63 sec, initial combustion occurred in the gas generator, and transient "bootstrap" operation was initiated. The higher gas generator chamber pressure observed during test 07, immediately after bootstrapping (Fig. 19d), was produced by increased fuel flow rate resulting from higher fuel system pressures.

The higher gas generator fuel injector pressure caused by higher fuel system resistance should result in a lower O/F ratio and lower the initial GGOT transient peak. This is confirmed by a comparison of initial peak GGOT values recorded for the two firings (Fig. 19d). The initial peak GGOT values were 1860 and 1650°F, respectively, for test 05 (cold thrust chamber) and test 07 (warm thrust chamber). It is concluded that the indicated lower initial peak value of GGOT for test 07 resulted from the higher fuel system resistance caused by the warm thrust chamber temperature conditions.

The initial GGOT peak was quenched by conditions resulting from thrust chamber ignition as discussed in Section 5.2.1.

5.2.2.2 Start Tank Energy Effect on GGOT, Initial Peak

The start tank energy effect can be demonstrated by comparing tests 09A and 10A. Test 09A was conducted with low start tank energy (2740 Btu), and test 10A was conducted using high start tank energy (2940 Btu) (Ref. 4). Thrust chamber and turbine hardware temperatures were similar for both tests.

Parameters that are affected by start tank energy and which may influence GGOT were investigated. The higher start tank energy resulted in a more rapid spinup of both turbines during start tank blowdown (Fig. 20a). Since the oxidizer valve was on the plateau and offered relatively high resistance to flow, the higher oxidizer turbine speed (with higher start tank energy) increased the oxidizer pump discharge and gas generator oxidizer injector pressures (Figs. 20b and c). Little change was observed in fuel pump discharge or gas generator fuel injector pressures (Figs. 20b and c). As a result, the higher gas generator oxidizer injector pressure caused by a higher start tank energy should result in an increase in O/F ratio and GGOT.

A comparison of the two tests indicated that the initial GGOT peaks were 1910 and 1700°F for high and low start tank energy, respectively (Fig. 20d). It is concluded that the indicated higher initial peak for GGOT during test 10A resulted primarily from high start tank energy effect on the gas generator system transient operation.

The initial GGOT peak was quenched by the conditions resulting from thrust chamber ignition, as discussed in Section 5.2.1.

5.2.2.3 Turbine Hardware Temperature Effect on GGOT, Initial Peak

The turbine hardware temperature (fuel turbine, crossover duct, and oxidizer turbine) is an indicator of energy available to the cold start tank gases during start tank blowdown. All tests, excluding 11B, were conducted with comparable turbine hardware temperatures (oxidizer turbine inlet temperature ranged from 20 to 53°F). No GGOT effects were observed from these small temperature variations. Firing 11B was conducted with warm turbine hardware temperature and an extremely cold thrust chamber. The turbine hardware temperature ranged from 88 to 157°F at ES over the length of the crossover duct with an oxidizer turbine inlet temperature of 134°F. The thrust chamber temperature was approximately -350°F at the thrust chamber throat and exit as a result of an 8-sec fuel lead. An excessive GGOT occurred during the start transient, and the firing was terminated early by ESCS. An extremely cold thrust chamber is a known contributor to excessive GGOT. However, warm turbine hardware is considered to be a contributor since it could impart a significant increase in start gas energy to the oxidizer turbine. The AS-203 Saturn IB flight indicated that the J-2 engine will be required to restart with warm turbine hardware temperature. Consequently, these temperature effects on GGOT should be evaluated in future tests.

5.2.2.4 Gas Generator Outlet Temperature, Second Peak

The second GGOT peak was observed on all tests and exceeded the initial peak on firings 9A, 10B, 11A, and 11B. The peak occurred approximately 1.3 sec after the STDV solenoid was energized. The second temperature peak is a function of the time required for the MOV second-stage ramp to begin, which in turn is a function of pressure differential across the valve (hydraulic torque). The hydraulic torque resists the effects of the valve pneumatic opening pressure and prolongs the time for the MOV second stage to begin opening (Fig. 21). The MOV offers high flow resistance until the second stage starts to open. This resistance causes initially high oxidizer pump discharge pressures and also a comparatively high oxidizer flow to the gas generator. Note in Fig. 18d that the oxidizer pump discharge pressure rise rate exceeded that of the fuel from 1.0 to 1.2 sec. The second stage of MOV started to open at 1.17 sec. Opening of the second stage permits a higher oxidizer flow to the thrust chamber; the combustion chamber pressure rises, and the resistance to fuel flow increases. This increased resistance causes the fuel pump discharge pressure rise rate to increase and converge on the oxidizer pump discharge pressure. An increased fuel flow is then supplied to the gas generator, which lowers the propellant mixture ratio and temperature. The effects of MOV second-stage opening time on GGOT second peak can be observed in Fig. 22. The temperature differential is GGOT second peak less the minimum after GGOT initial peak. The temperature differential increased with MOV second-stage initial opening time. Conditions that would adversely affect the second temperature peak would be an unusually high energy supply to the oxidizer turbopump during start transient, which would cause a prolonged oxidizer valve plateau time and a higher oxidizer flow to the gas generator. The MOV plateau times and values of the GGOT second peaks for all tests are presented in Table V.

The maximum GGOT second peak occurred during firing 11B (Fig. 23). The overshoot of 2150°F exceeded the automatic "kill" limit, and the firing was terminated by the engine safety cutoff logic. Start conditions existing for test 11B were (1) high start tank energy, (2) warm turbine hardware temperature (ranged from 88 to 157°F over the crossover duct length), and (3) a cold thrust chamber (-350°F) that resulted from an 8-sec fuel lead.

5.2.3 Thrust Chamber Side Forces

One of the primary objectives of this test program was to verify that the pitch and yaw side forces produced during the ES transient at altitude conditions were within vehicle specifications. The J-2 engines exhibit side forces in excess of 20,000 lb_f during sea-level ES which are attributed to unsymmetrical flow separation in the thrust chamber. Minimal side forces were anticipated at ES at altitude conditions, since nozzle full flow transition occurs at thrust chamber ignition and with considerably lower pressures in transient flow separation regions.

Figure 24 presents pitch and yaw force data which are typical in magnitude of the engine side forces measured during the AEDC tests. The natural frequency of the side-force measuring system (approximately 20-cps) superimposed on the engine side-force data was eliminated from the data presented, in order to show the side forces produced by the engine. The force measuring system interaction tolerance was ±200 lb_f.

Pitch and yaw forces recorded during the AEDC tests indicate that the side forces produced by the J-2 engine at pressure altitude conditions are significantly less than acceptance test values and well within acceptable limits for the J-2 engine Saturn IB and Saturn V flight applications.

5.2.4 Combustion Instability

Analysis of engine vibration (oxidizer injector dome) and combustion chamber pressure data indicated instability during thrust chamber ignition on all engine firings except tests 06 and 10B. Chamber pressure fluctuations at frequencies of approximately 300 cps were recorded. The chamber pressure fluctuation periods correlated in time with the instability observed on the vibration data. Combustion instability was not anticipated during the AEDC tests.

The duration of the instability periods varied from 3 to 132 msec for ten of the applicable engine firings. Instability was not observed on tests 06 and 10B. Main chamber ignition was not obtained, as planned, for tests 9B and 11C. Test 04 resulted in an engine automatic VSC cutoff after 112 msec of instability; the instability continued for 20 msec after engine cutoff had been initiated. Major blockage of the injector was considered as a possible contributor to the instability. However, instrumentation was inadequate for evaluating the injector oxidizer flow resistance during start transient. Consequently, there are no conclusive data to support oxidizer post blockage during test 04 as a source of recorded instability. The variations in combustion chamber pressure for tests with instability periods of 0, 25, 67, and 132 msec are presented in Fig. 25.

Comparisons of J-2 engine vibration data from the Saturn IB AS-201 through -203 flights and AEDC tests revealed vibration similarity. The predominant vibration frequency recorded during the instability periods for Saturn IB flights and AEDC tests was 2100 cps. This frequency corresponds to the calculated first tangential mode of vibration of the combustion chamber. The amplitude of the vibration frequency recorded during flight was about a factor of two less than that observed during AEDC tests. The duration of the instability during flight ranged from 30 to 60 msec.

The instabilities experienced during the AEDC tests are believed to be real and representative of the J-2 ES transient operation at altitude conditions. An acceptable relationship correlating the dynamics producing the undesirable instability could not be developed from the data collected. Instrumentation for such transient analysis of engine operation was not provided on the engine.

5.2.5 Transient Fuel Pump Performance

Transient fuel pump head/flow data from all firings were documented and compared with stall inception design data. Fuel system resistance was never high enough to cause a pump stall, but the limiting value was more nearly approached on test 07 (Fig. 26) than any other test. The higher fuel system resistance to flow, resulting from warm thrust chamber temperature, resulted in an increase in pump blade loading, thereby causing the indicated approach toward the stall limit. A more representative fuel pump head/flow characteristic can be observed from test 05 data (Fig. 26). Thrust chamber temperatures at ES dictating the resistance to fuel flow for these tests were as follows:

Test No.	Throat Temperatures, °F	Exit Temperatures, °F	Resistance to Fuel Flow
5	-233	-216	Low
7	-88	-70	High

The warm thrust chamber temperature conditions experienced at ES for test 07 would have produced, in all probability, a low level fuel pump stall in a sea-level J-2 engine firing. Therefore, the warmest thrust chamber temperature conditions specified for a safe engine start by the engine manufacturer (i. e., a fuel injector temperature no warmer than -150°F) could conceivably be revised to a warmer temperature limit for altitude testing and flight.

5.2.6 Fuel Lead Effects

A 1-sec fuel lead preceded all tests except 9B, 10B, and 11B. A fuel lead of 8 sec was completed before tests 9B and 11B and of 4.5 sec before test 10B. Thrust chamber pre-fire temperature conditioning was not conducted before 4.5- and 8-sec fuel leads but was accomplished before 1-sec fuel lead tests.

The effects of 4.5-sec (test 10B) and 8.0-sec (test 9B) fuel leads on thrust chamber temperature are shown in Fig. 27a. For both tests, the initial ambient thrust chamber temperature was between 25 and 50°F. The average thrust chamber temperature for the two tests, after 4.5 sec of fuel lead, was -240°F at the exit, -118°F at the throat, and -211°F at the fuel injector. After completion of the 8-sec fuel lead, the exit temperature was -296°F, the throat temperature was -245°F, and the fuel injector temperature was -420°F.

The fuel lead more effectively chills the thrust chamber at altitude than at sea-level pressures. The major differences observed at altitude conditions were (1) a lower backpressure or resistance to fuel flow, (2) sonic flow through the injector and throat, and (3) less heat transfer to the thrust chamber at altitude (at sea level, additional heat is added by a water-cooled thrust chamber diffuser and recirculation of hot gas inside the thrust chamber). Sea-level acceptance and AEDC altitude data indicate fuel injector temperatures of -165 and -420°F, respectively, after a fuel lead of approximately 8 sec (Fig. 27a). The fuel injector temperature before the fuel lead at sea level was 42°F.

An indication of the fuel system resistance drop as a function of temperature drop can be obtained from Fig. 27b (presented to indicate trends only). Note that after approximately 3.0 sec, the fuel flow exhibits an increasing trend as the pressure differential decreases.

Slight thrust chamber temperature drops were noted after 1-sec fuel leads with the prechilled thrust chamber. Average temperature drops were 70°F at the exit and 25°F at the fuel injector.

5.2.7 Engine Shutdown Transient

Selected shutdown transient traces of typical engine valve and pressure parameters, obtained from high response oscillograph data, are presented in Fig. 28. Valve operating times at shutdown for all firings are depicted in Table X. Valve operating times were within specified limits in all cases. Engine shutdown transients were consistent and satisfactory in all cases, and a detailed transient discussion is omitted.

5.3 PERFORMANCE

Performance data were calculated from test measurements utilizing the PAST 640 computer program, a standard J-2 engine performance program developed and programmed by the engine manufacturer. This program calculates the engine and engine component performance based on (1) measured data or "site" and (2) measured data normalized to standard pump inlet engine ambient vacuum conditions. The required program constants, which included engine dimensional measurements, engine flowmeter calibration constants, pump head/flow constants, pump efficiency constants, and thrust coefficient curve fit constants, were provided by the engine manufacturer. Engine test measurements required by the performance program are itemized in Table XI. These measurements were obtained from the digital data acquisition system by averaging the 40 data samples obtained in the 1-sec time intervals of interest for each measured parameter required for the program. Fuel and oxidizer engine flowmeter cyclic output data were manually reduced from oscillograph.

Selected engine performance data computed for site conditions are presented in Table VI. The short duration of the engine firings during these tests limits the utility of the performance data. The original program plan was to conduct a 60-sec engine firing to permit attainment of engine thermal equilibrium for valid comparison of AEDC and acceptance test results. However, a 60-sec engine firing was not accomplished on the tests reported herein. Engine firing durations ranged from 5 to 30 sec for tests with an engine O/F ratio of 5.0. A 40-sec duration firing was conducted with a 5.5 engine O/F ratio.

The basic engine performance at AEDC is depicted in Fig. 29. It is observed that the thrust chamber fuel and oxidizer flow rates were slightly lower than nominal; however, the gas generator fuel and oxidizer flow rates were below the minimum requirements. The thrust chamber and gas generator propellant flow rates during these tests compared closely but were consistently lower than those values recorded during engine acceptance tests. The curves classifying the flows as maximum, nominal, or minimum represent a block of J-2 engines (Ref. 2).

Factors limiting the confidence level in the engine performance data presented are (1) engine thrust was not measured but was calculated from a chamber pressure relationship established during acceptance tests, and (2) redundant propellant flowmeters were not used.

The S-IVB battleship stage performance was consistent and satisfactory for each test. Performance data for selected stage parameters are summarized in Table VII.

5.4 PROPELLANT DRY DUCT/PUMP INLET CONDITIONING

Three tests, not involving an engine firing, were conducted to simulate and evaluate the planned method for temperature conditioning the hardware associated with the propellant low pressure ducts and turbopumps for the J-2 engine restart application during the

Saturn V/Apollo flights. These tests, conducted during the 09, 10, and 11 test periods, involved chilldown of the said hardware, from an initially dry condition, using the stage propellant recirculation systems with the propellant prevalves closed. The propellant recirculation systems circulate propellants (fuel at about 143 gpm, oxidizer at about 43 gpm) from the vehicle propellant tanks, through the propellant low pressure ducts and turbopumps, and back into the vehicle propellant tanks (Fig. 9). Two of the tests consisted of a 5-min recirculation period; the third test consisted of a 10-min recirculation period.

Results and conclusions drawn from these tests are as follows:

1. A 5-min recirculation period was inadequate to properly temperature condition the fuel low pressure duct and turbopump hardware. At the termination of the recirculation period — the opening of the propellant prevalves in preparation for engine restart — the fuel pump inlet conditions remained in the safe ES region for only approximately 3 sec before returning to saturation conditions. However, the oxidizer pump inlet conditions remained in the safe ES region for an indefinite time period.
2. The 10-min recirculation period, including cycling the prevalves, provided an adequate means for temperature conditioning the hardware of both propellant low pressure ducts and turbopumps. With the opening of the prevalves at the conclusion of the recirculation period, the fuel and oxidizer pump inlet conditions remained well within the respective safe ES regions. An inadvertent procedural error, resulting in the prevalves being cycled opened for approximately 13 sec, was encountered 10 sec after the chilldown recirculation had been initiated. Momentary cycling of the prevalves in the initial phase of the chilldown has been concluded to be an useful event in the chilldown cycle.
3. Saturn IB flight data indicate the low pressure ducts and pumps are not void of propellants before restart. Therefore, the AEDC tests present "worse-case"-type chilldown results.

5.5 FUEL START TANK REPRESSURIZATION AND WARMUP

The H_2 sources for start tank repressurization are provided by tapoffs at the fuel manifold inlet and the fuel injector (Fig. 15). Once the start tank is pressurized to a level equivalent to the injector pressure (approximately 5 sec after the STDV solenoid is energized), the repressurization source is limited to the manifold inlet supply. The maximum start tank pressure attainable during a firing is limited by the fuel manifold inlet pressure and firing duration.

Fuel start tank repressurization data from five firings are shown in Fig. 30a; duration of the firings presented ranged from 5 to 40 sec. Start tank pressures at engine shutdown ranged from 746 psia after the 5-sec firing (test 9A) to 1056 psia after the 40-sec firing (test 11A). Respective temperatures at shutdown of the 5- and 40-sec firing were -223 and -262°F.

The fuel start tank must contain at least 2.5 lb_m (Fig. 30b) of GH_2 to meet the minimum start tank energy requirements. The mass existing at shutdown for the firings was determined and indicated that a firing duration greater than 20 sec is necessary to provide the minimum mass required to meet the minimum restart start tank energy. After the 20-sec firing, the calculated mass was 2.32 lb_m, and after 30 sec, the mass was 2.88 lb_m.

After shutdown of a 40-sec firing, the start tank conditions warmed to the safe start region in approximately 7 min. Start tank conditions existing at 10-min increments up to 50 min after shutdown are shown in Fig. 30b. A temperature-sensitive relief valve kept conditions from exceeding structural limits of the start tank. From the warmup data presented, the relief valve was observed to become active at approximately 1350 psia and -203°F.

Warmup data from the Saturn IB AS-203 flight are also shown in Fig. 30b. Conditions existing at engine cutoff and end of first, second, third, and fourth orbit are noted.

5.6 THRUST CHAMBER TEMPERATURE CONDITIONING

Pre-fire temperature conditioning of the thrust chamber was necessary to meet the conditions established for ES following the boost-phase warmup during launch. During these tests, the boost-phase warmup was not simulated because of facility test cell purge requirements. Temperature conditioning was accomplished by circulating He through a facility LH_2 heat exchanger to obtain cold He, which was supplied to engine customer connect and then circulated through the thrust chamber.

A method was developed early in the program that made it possible to obtain any desired customer connect temperature. With the capability of prechilling to any desired temperatures, and the repeatability of short coast temperature rises, temperature requirements were met. Thrust chamber preconditioning data obtained during a typical test (test 10A) are presented in Fig. 31. The conditions existing 220 sec before STDV solenoid energized are shown; at this time, throat temperatures were constant at -250°F and exit temperature of -265°F. Predictable temperature changes caused by initiation of maximum cell inerting purges (of approximately 30 lb_m/sec) and facility steam ejector operation made it possible to obtain desired values at ES. After terminating the prechill, exit thermocouples warmed to approximately -166°F and stabilized. Thrust chamber skin temperature at the throat warmed to approximately -177°F and was essentially stable. The throat temperature, as indicated by TTC-1P, exhibited a continual increase after termination of the prechill. At t_o - 60 sec, a rise occurred in the exit temperature. This rise was caused by some secondary cell flow and does not represent typical warmup. The curve is extrapolated in this region to show the expected data.

SECTION VI
SUMMARY OF RESULTS

The results of the 14 tests of the J-2 rocket engine (in the Saturn S-IVB stage configuration) conducted at simulated pressure altitude conditions in Test Cell J-4 are summarized as follows:

1. Altitude testing of the J-2 engine revealed that start transient characteristics at AEDC were different from the engine acceptance tests and provided an important insight into the J-2 engine transient operation for the Saturn IB and Saturn V flights.

2. Tests at AEDC demonstrated that start characteristics are primarily a function of (a) thrust chamber temperature conditions at ES, (b) start energy delivered to the turbines, (c) MOV opening sequence, and (d) ambient pressure altitude conditions.

3. Transient GGOT were consistently higher than engine acceptance tests and resulted in a premature termination of one test. The higher transient temperatures were attributed to improved oxidizer quality to the gas generator and lower fuel system resistance at altitude conditions.

4. The GGOT exhibited two transient peaks. The initial peak was demonstrated to be a function of thrust chamber conditioning temperature and start tank energy. The second peak was

demonstrated to be a function of the MOV operating sequence. Additional testing is necessary to predict the effect of turbine hardware temperature on GGOT.

5. Combustion instability during engine ignition was observed for time durations ranging from 3 to 132 msec for ten of the engine firings and compared favorably with the instability recorded during the Saturn IB AS-201 through -203 flights.

6. Engine transient side forces measured at AEDC were significantly less than those recorded during sea-level engine acceptance tests and are well within acceptable limits.

7. Fuel leads at altitude conditions were much more efficient in thrust chamber chilldown than at sea level.

8. Shutdown transients were satisfactory, and all valve operating times were within limits.

9. A 10-min propellant recirculation period, including cycling the prevalves, provided an adequate means for temperature conditioning the hardware of both propellant low pressure ducts and turbopumps from an initially dry condition for safe engine restart.

10. Engine firing durations greater than 20 sec are required to recharge the start tank to the minimum mass level required for safe engine restart.

REFERENCES

1. Test Facilities Handbook, (6th Edition). "Large Rocket Facility, Vol. 3." Arnold Engineering Development Center, November 1966.

2. "J-2 Rocket Engine, Technical Manual Engine Data." R-3825-1, August 1965.

3. Dubin, M., Sissenwine, N., and Wexler, H. "U. S. Standard Atmosphere, 1962." December 1962.

4. "Tables of Thermal Properties of Gases." National Bureau of Standards Circular 564, November 1965.

APPENDIXES

I. ILLUSTRATIONS

II. TABLES

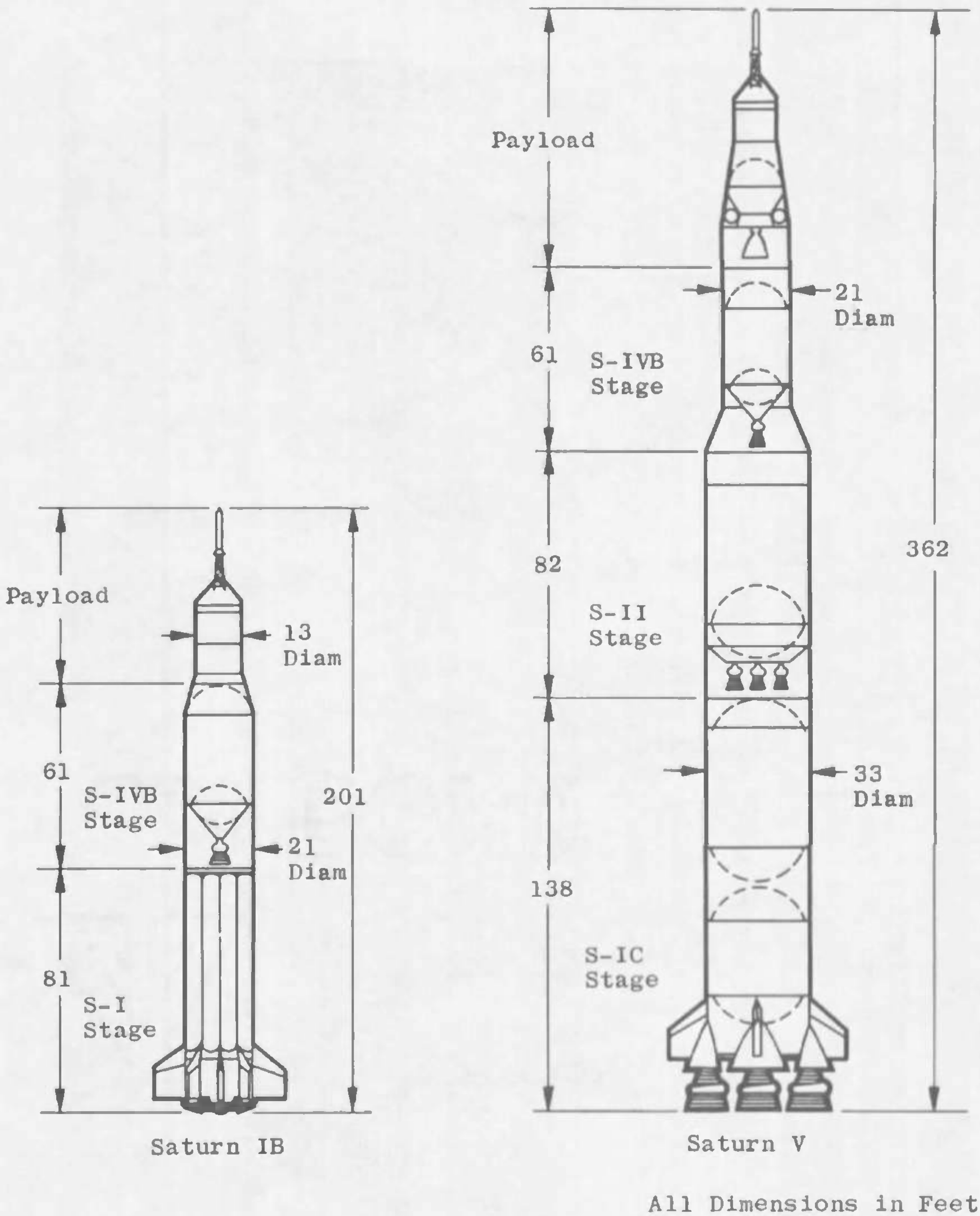
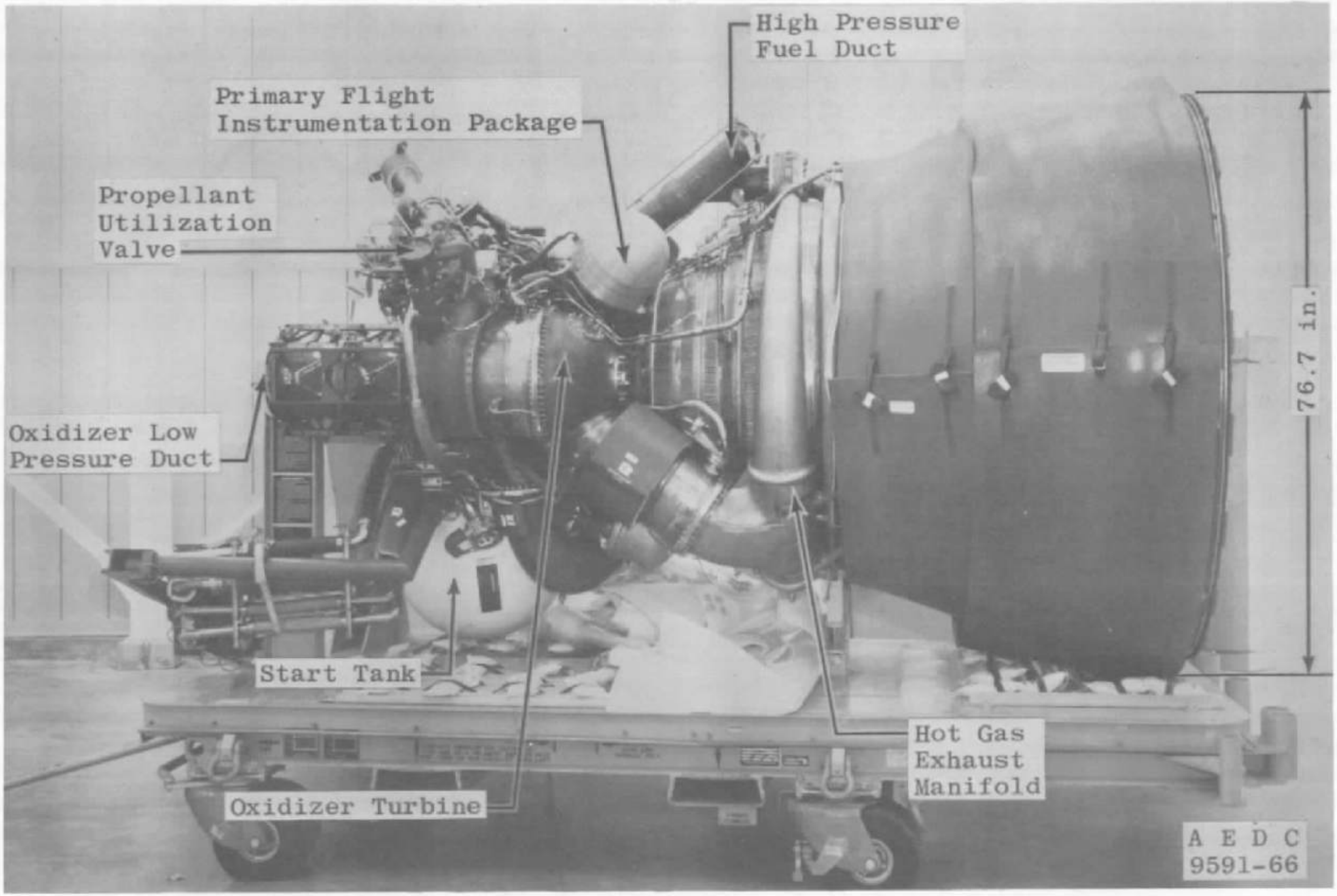
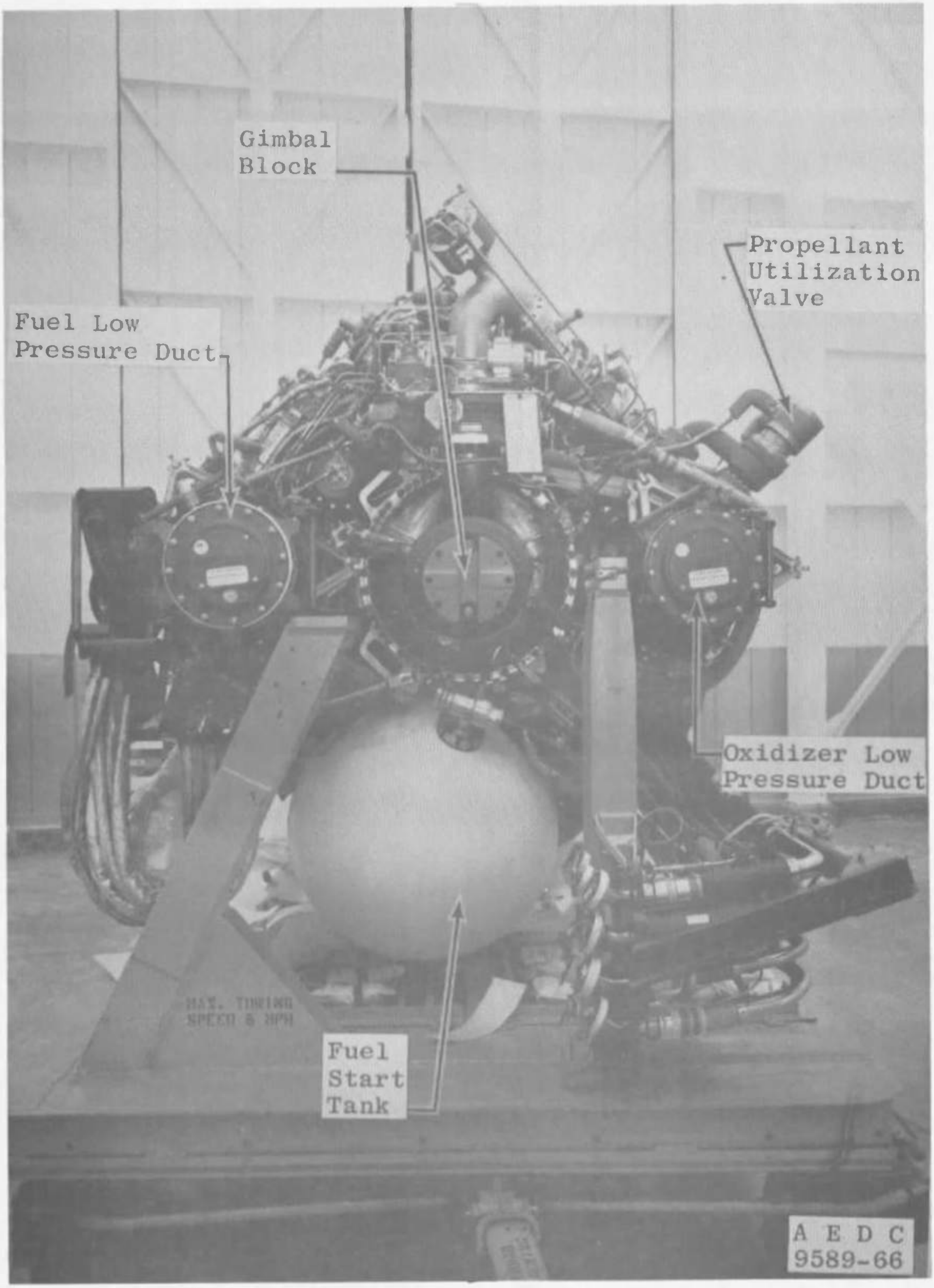


Fig. 1 Saturn S-IB and S-V Launch Vehicle



a. Side View

Fig. 2 J-2 Engine, S/N 2052



b. Top View

Fig. 2 Concluded

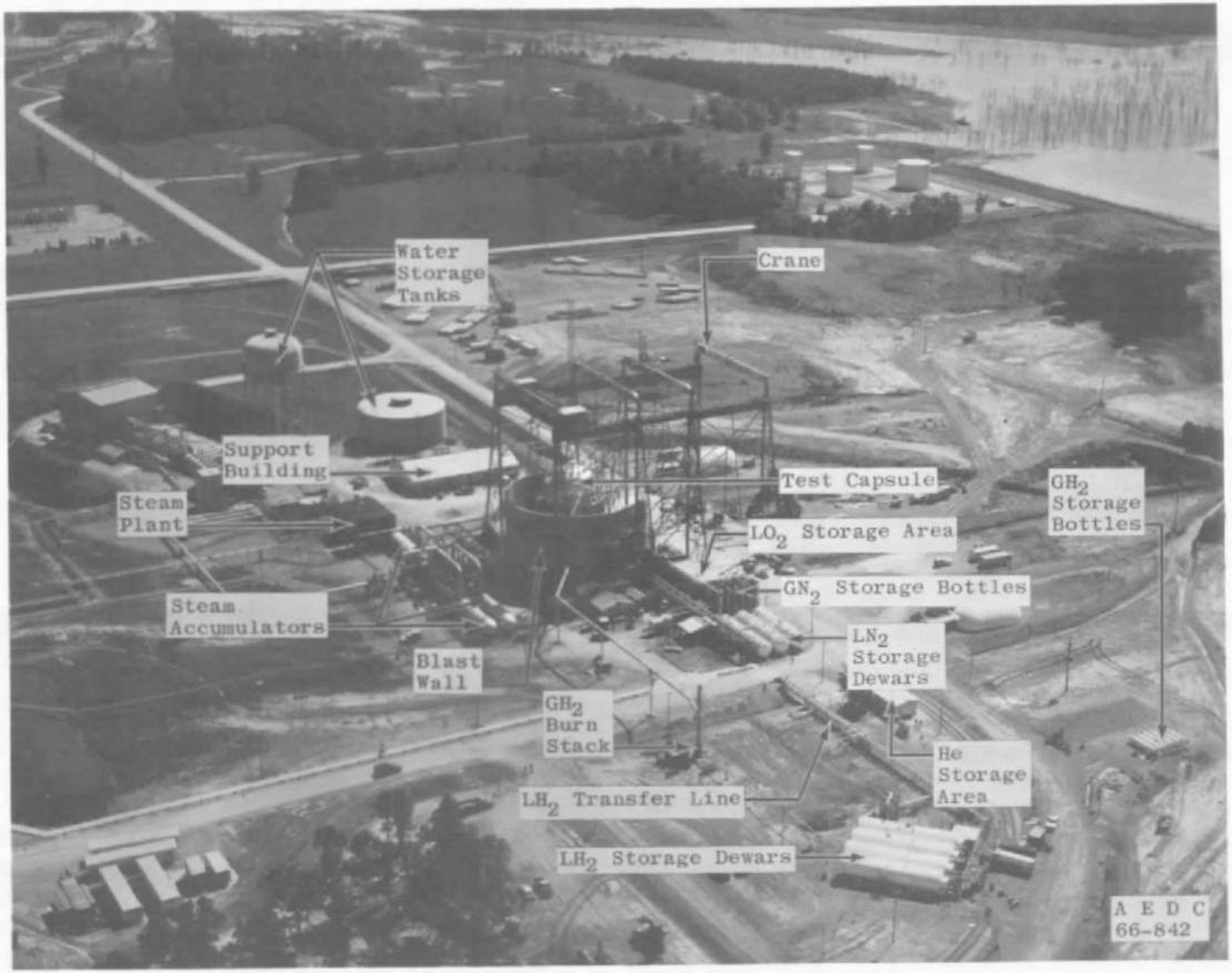


Fig. 3 Aerial View of Test Cell J-4

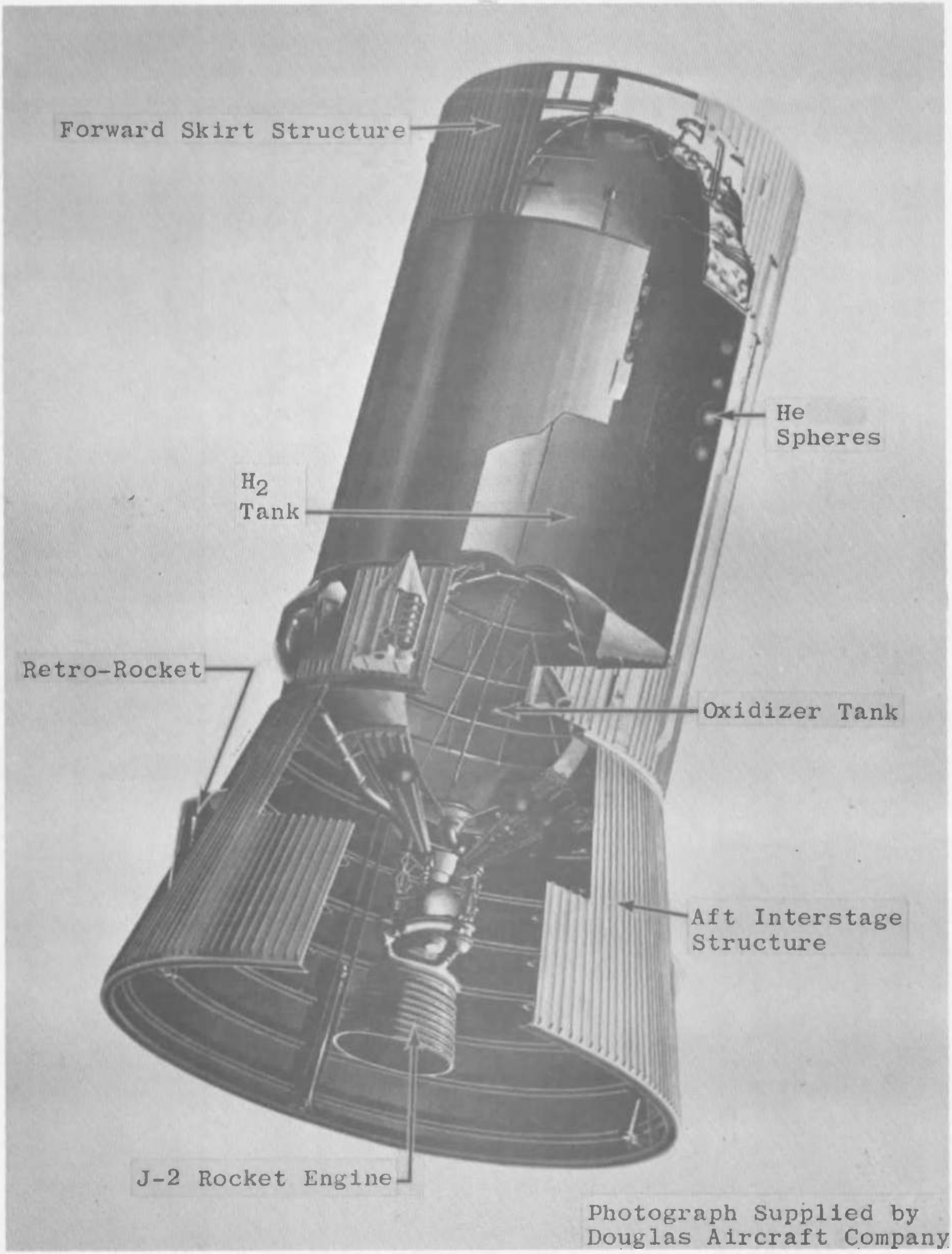


Fig. 4 J-2 Engine and S-IVB Flight Stage Assembly

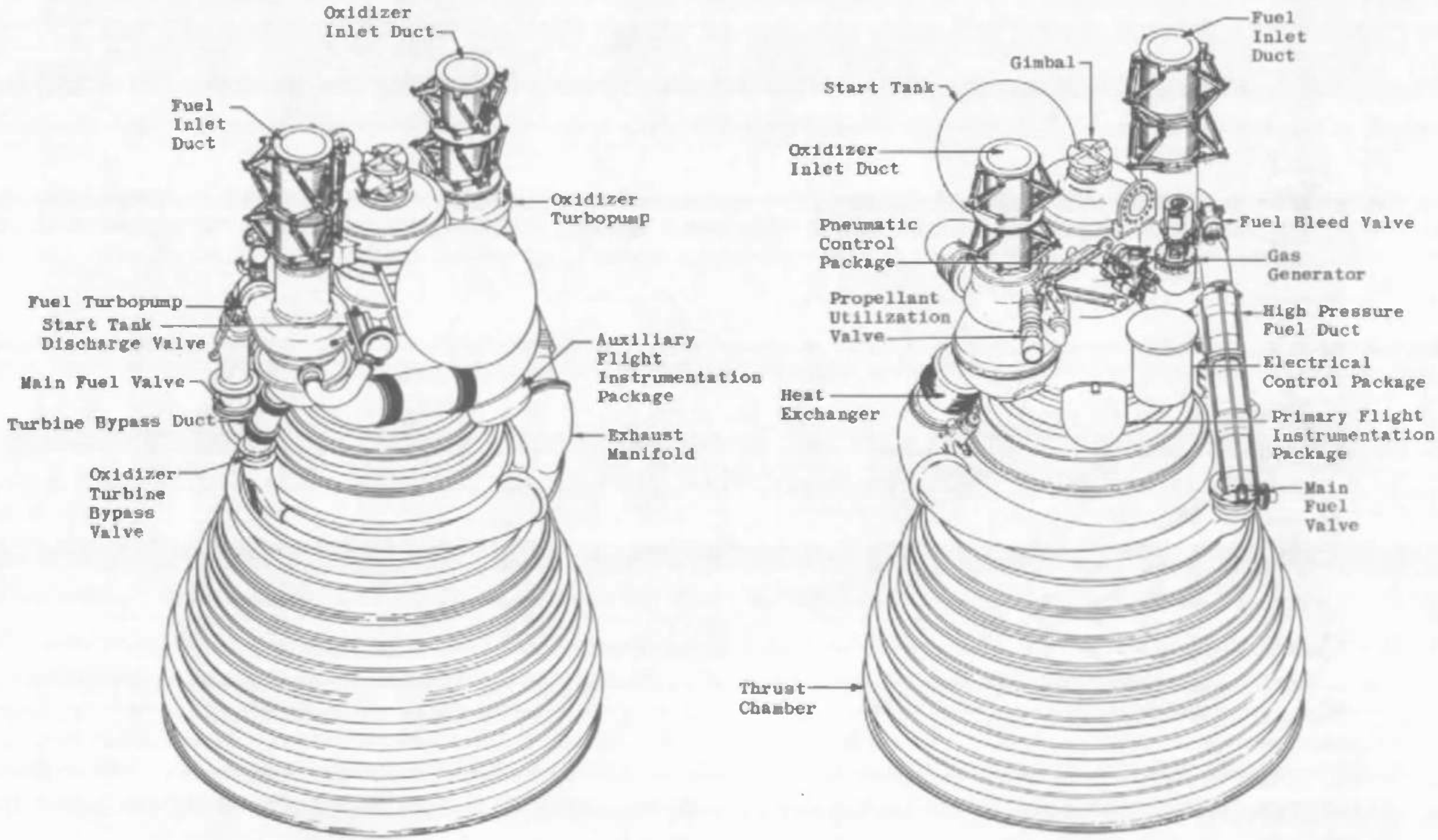


Fig. 5 Details of the J-2 Engine

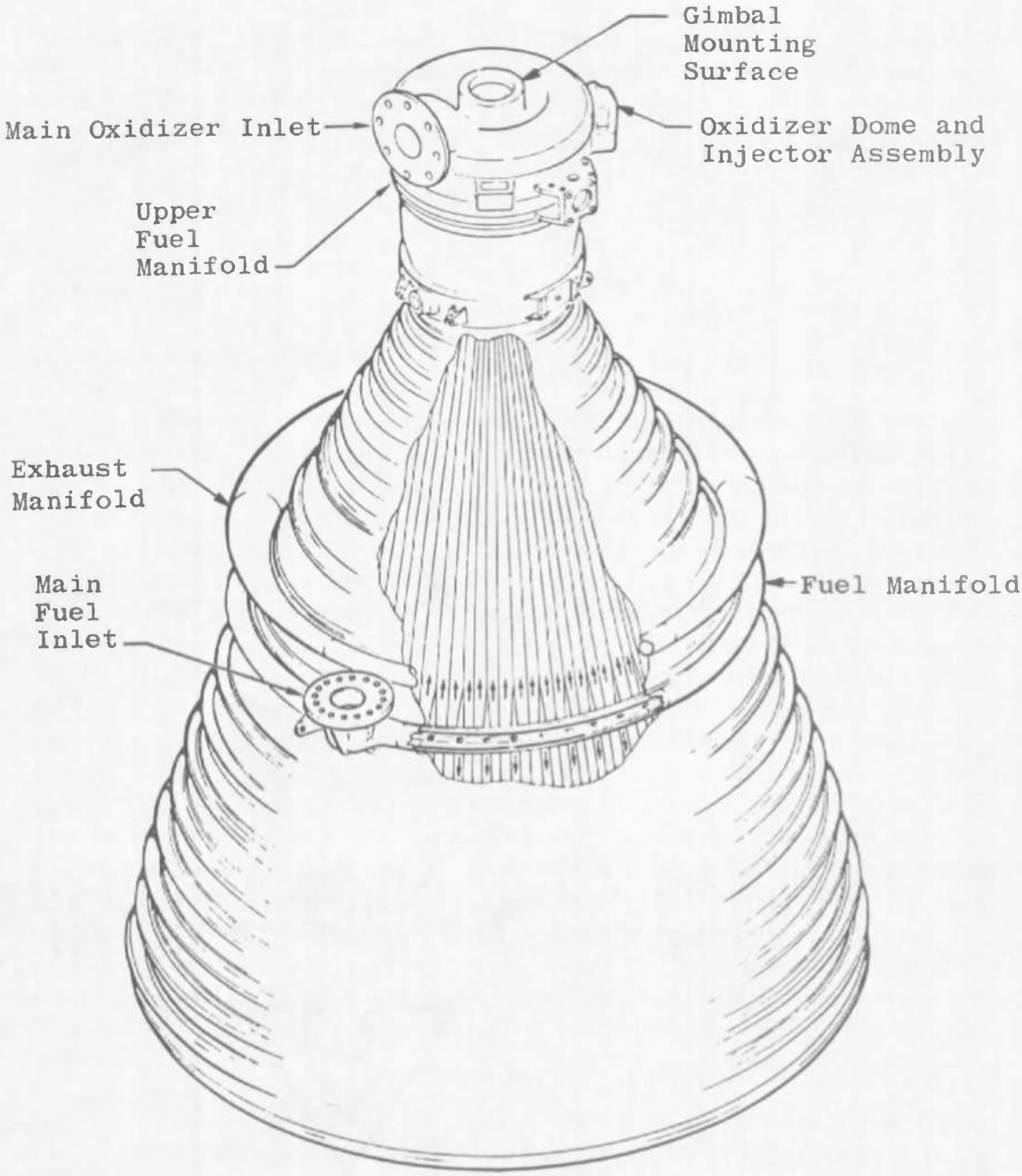


Fig. 6 Details of the J-2 Engine Thrust Chamber

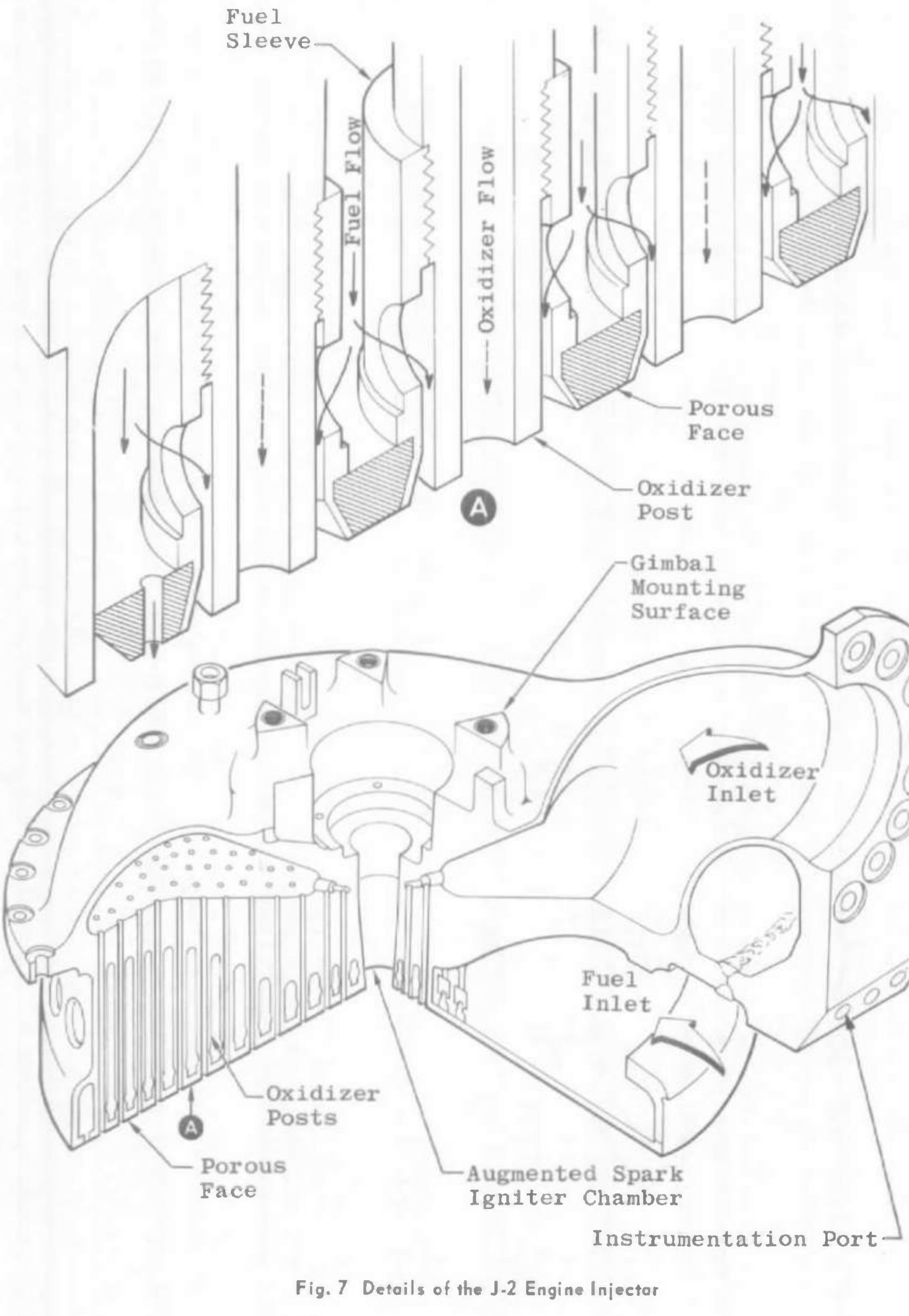


Fig. 7 Details of the J-2 Engine Injector

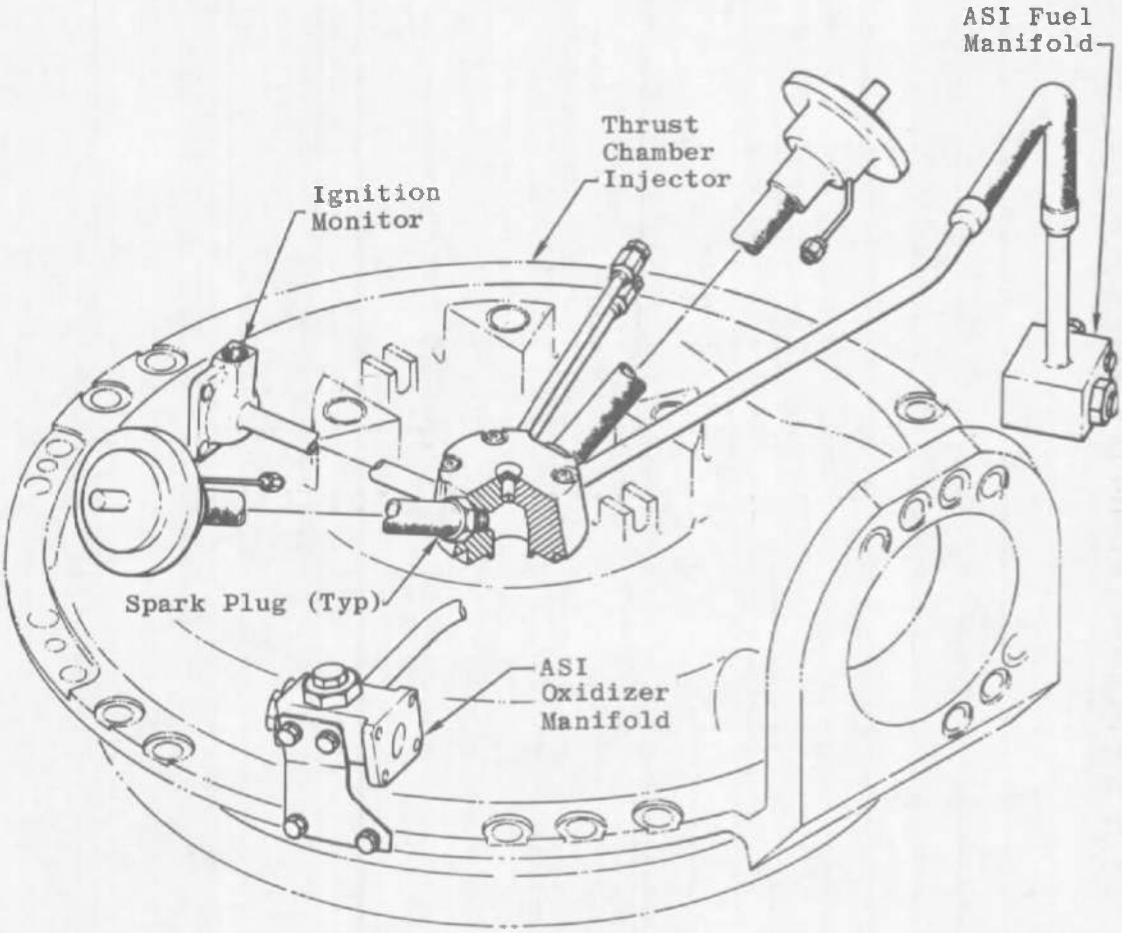


Fig. 8 Details of the Augmented Spark Igniter

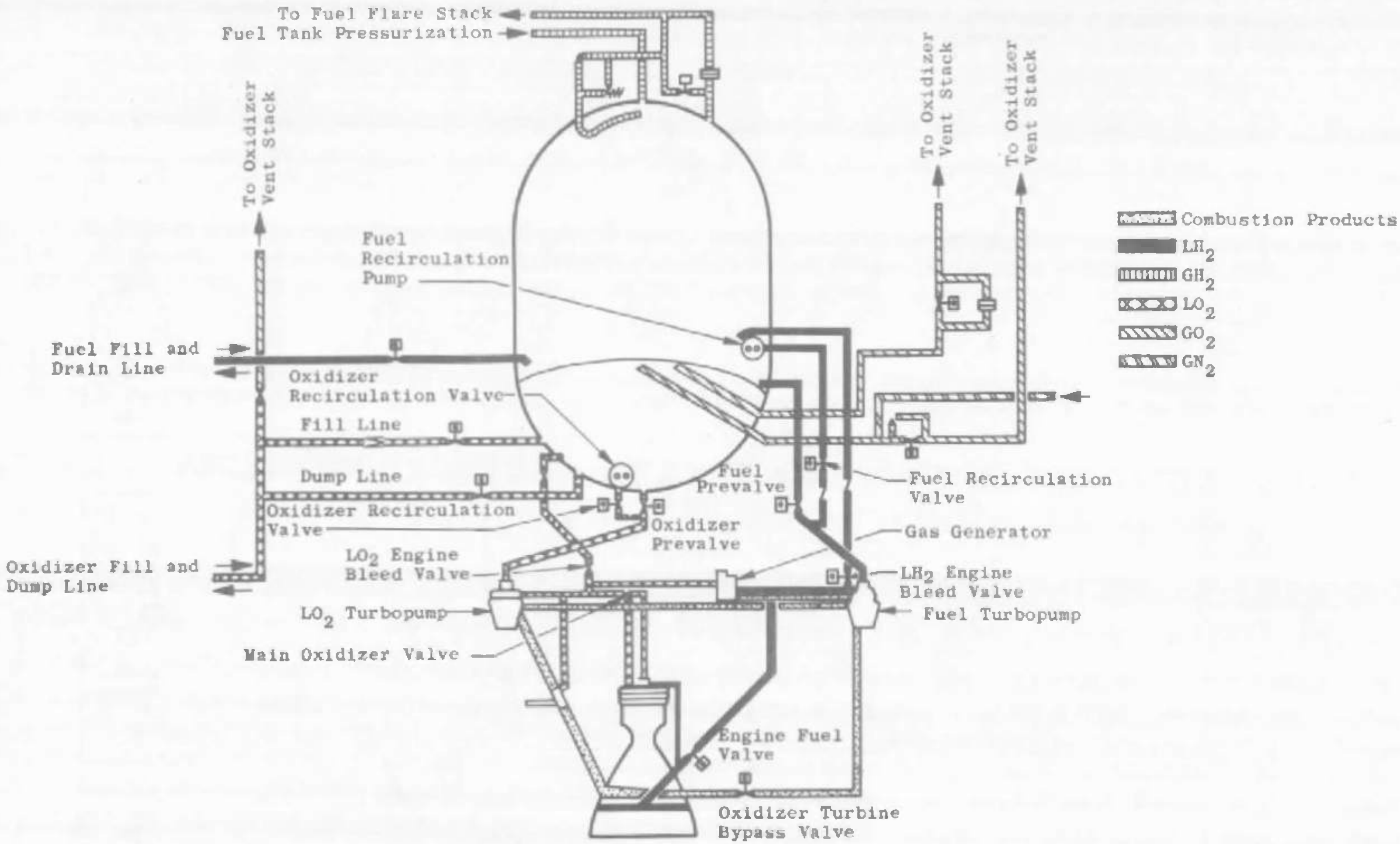


Fig. 9 Schematic of the S-IVB Stage/J-2 Engine

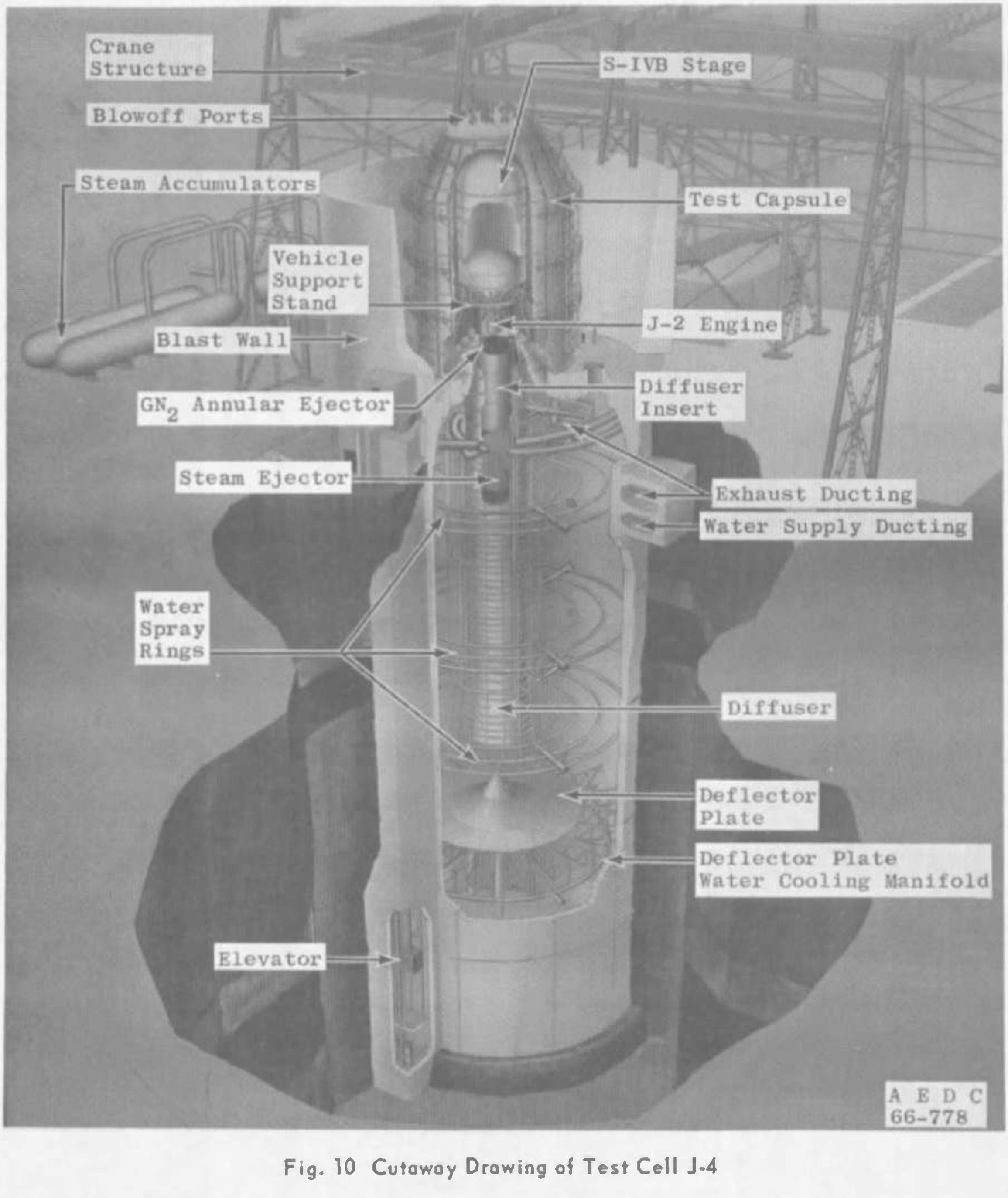


Fig. 10 Cutaway Drawing of Test Cell J-4

Identification Number	Consumable System	Additional Capacity	Current Capacity
1	LH_2	112,000 gal	112,000 gal
2	LN_2	84,000 gal	88,630 gal
3	GN_2	1,311,000 scf[1]	1,720,000 scf[1]
4	GH_2	420,000 scf[1]	420,000 scf[1]
5	GHe_2	401,000 scf[1]	401,000 scf[1]
6	Steam	450,000 lb_m	1,350,000 lb_m
7	LO_2	---	20,000 gal
8	Water	---	1,750,000 gal

[1]Standard cubic feet at 5000 psia

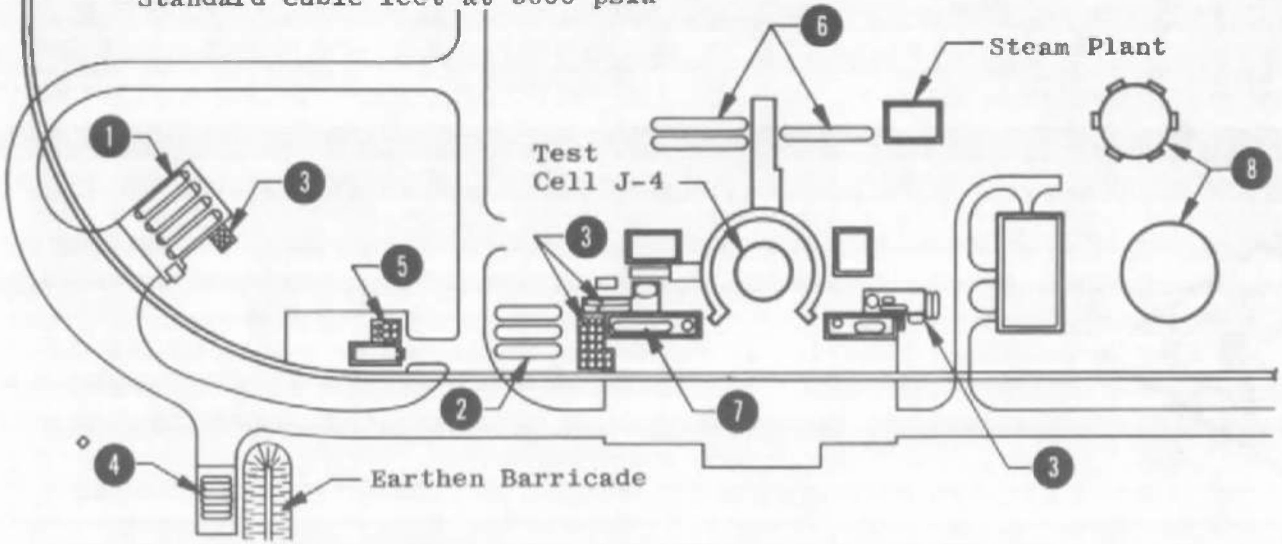
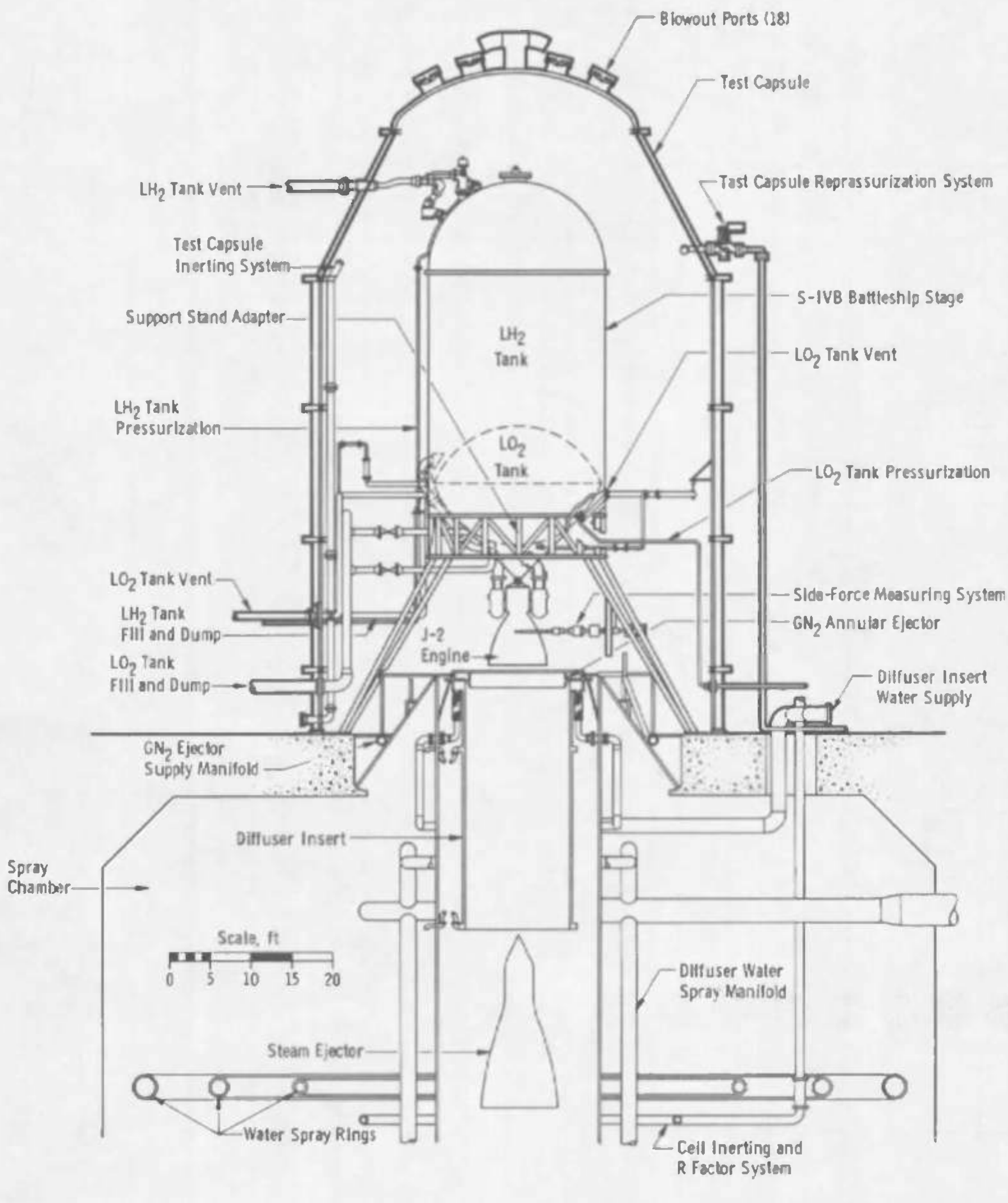
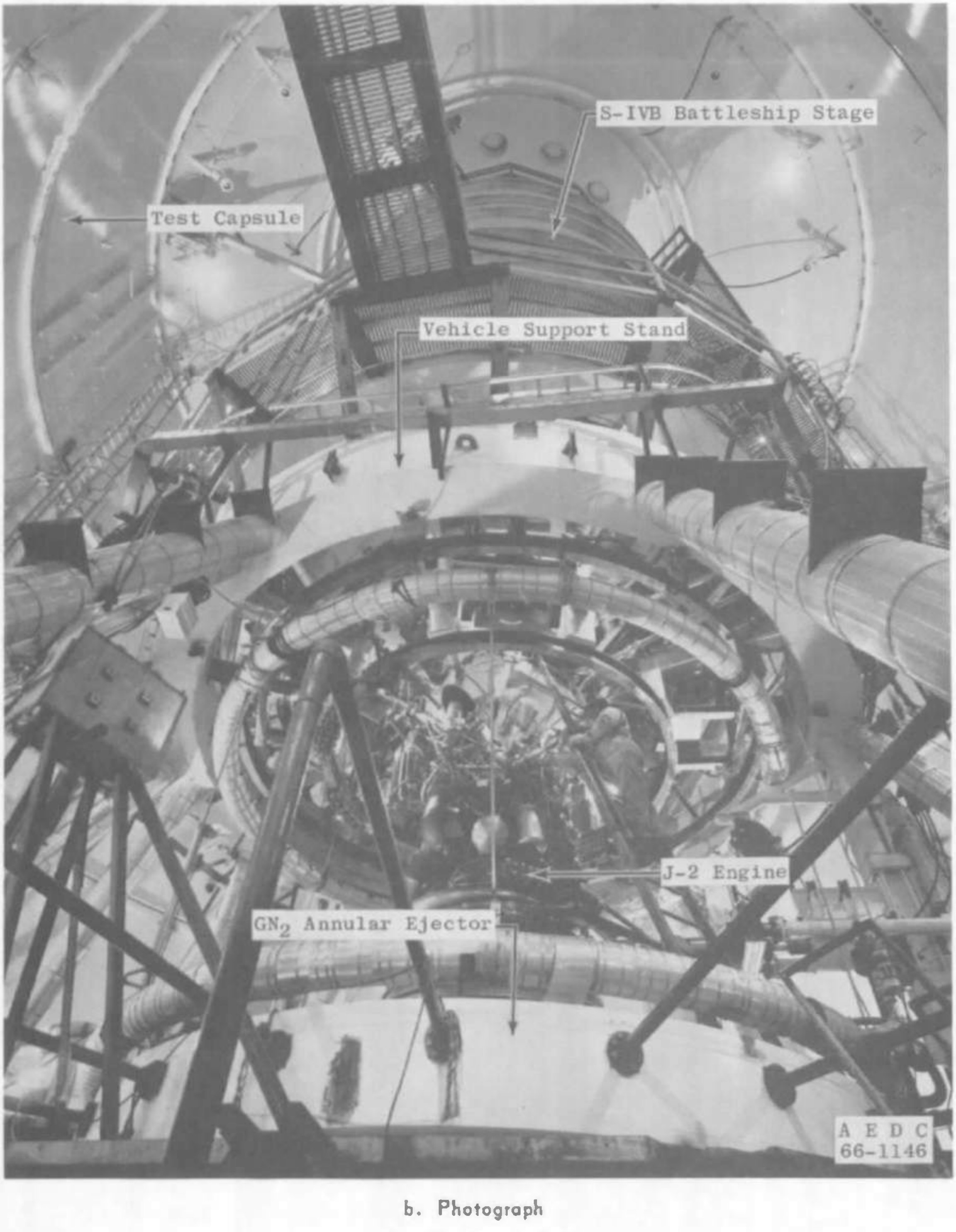


Fig. 11 Test Cell J-4 Consumable Storage Systems, Plan View



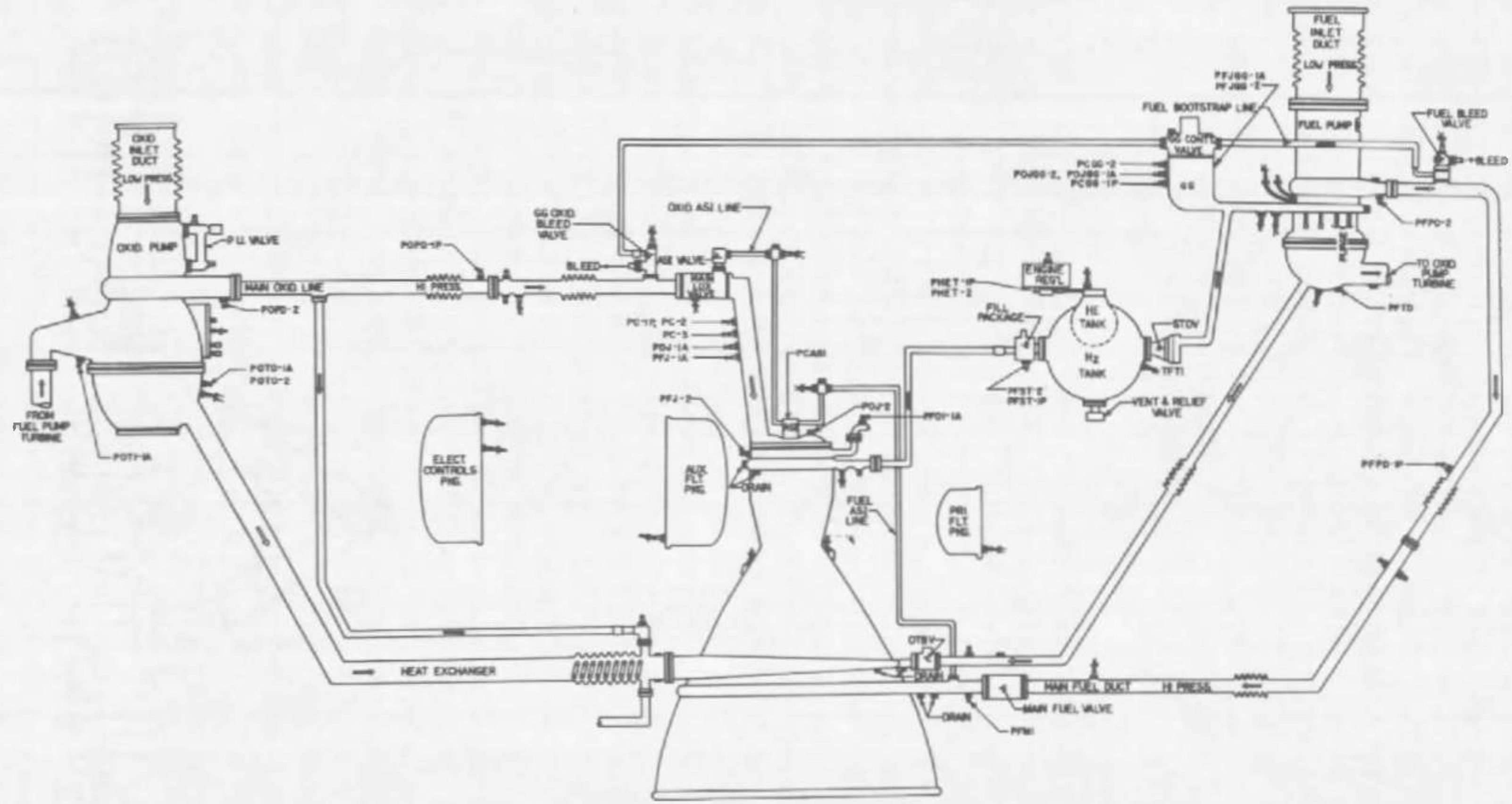
a. Elevation View

Fig. 12 Installation of the J-2 Engine/S-IVB Battleship Stage in Test Cell J-4



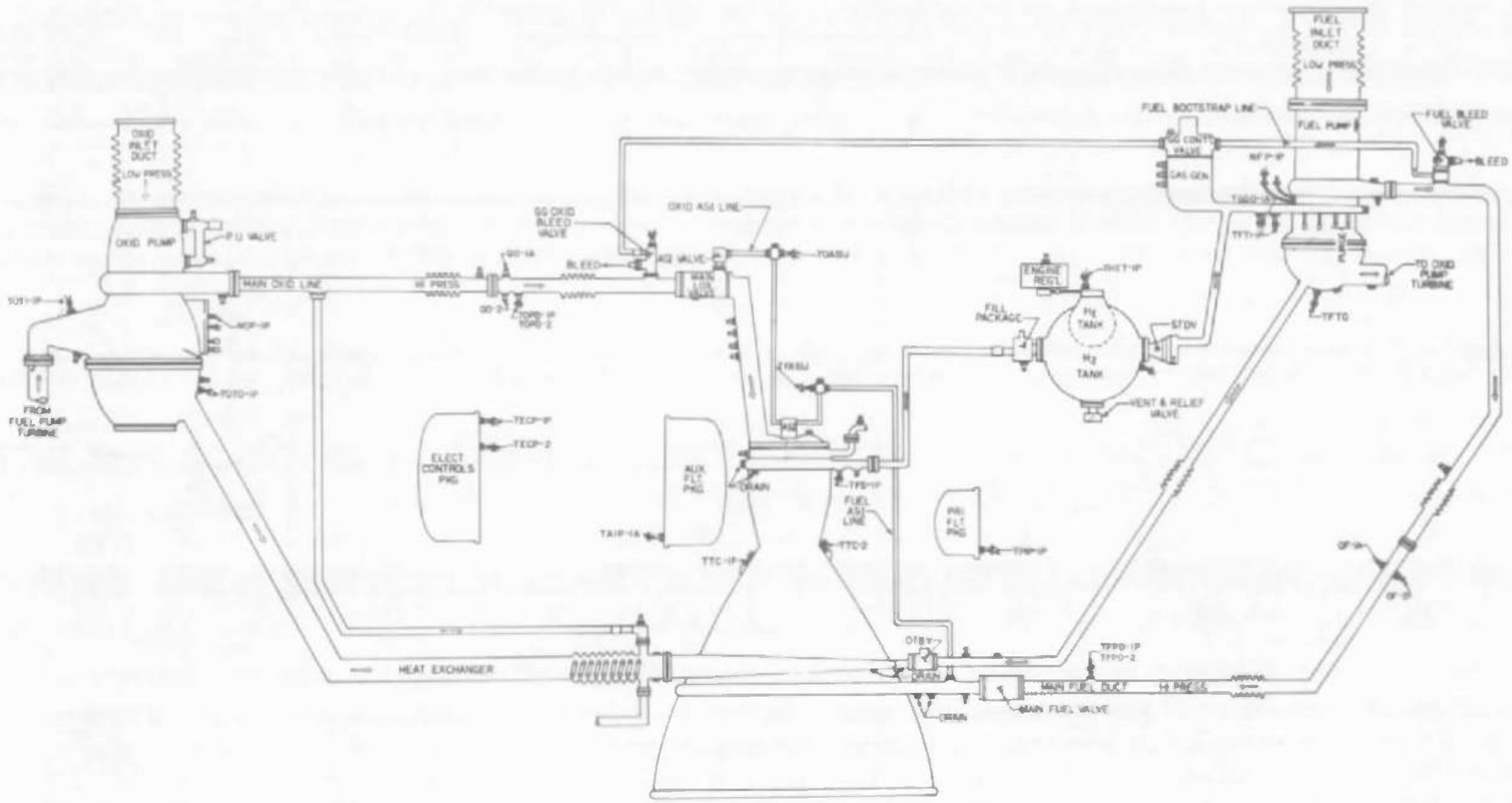
b. Photograph

Fig. 12 Concluded



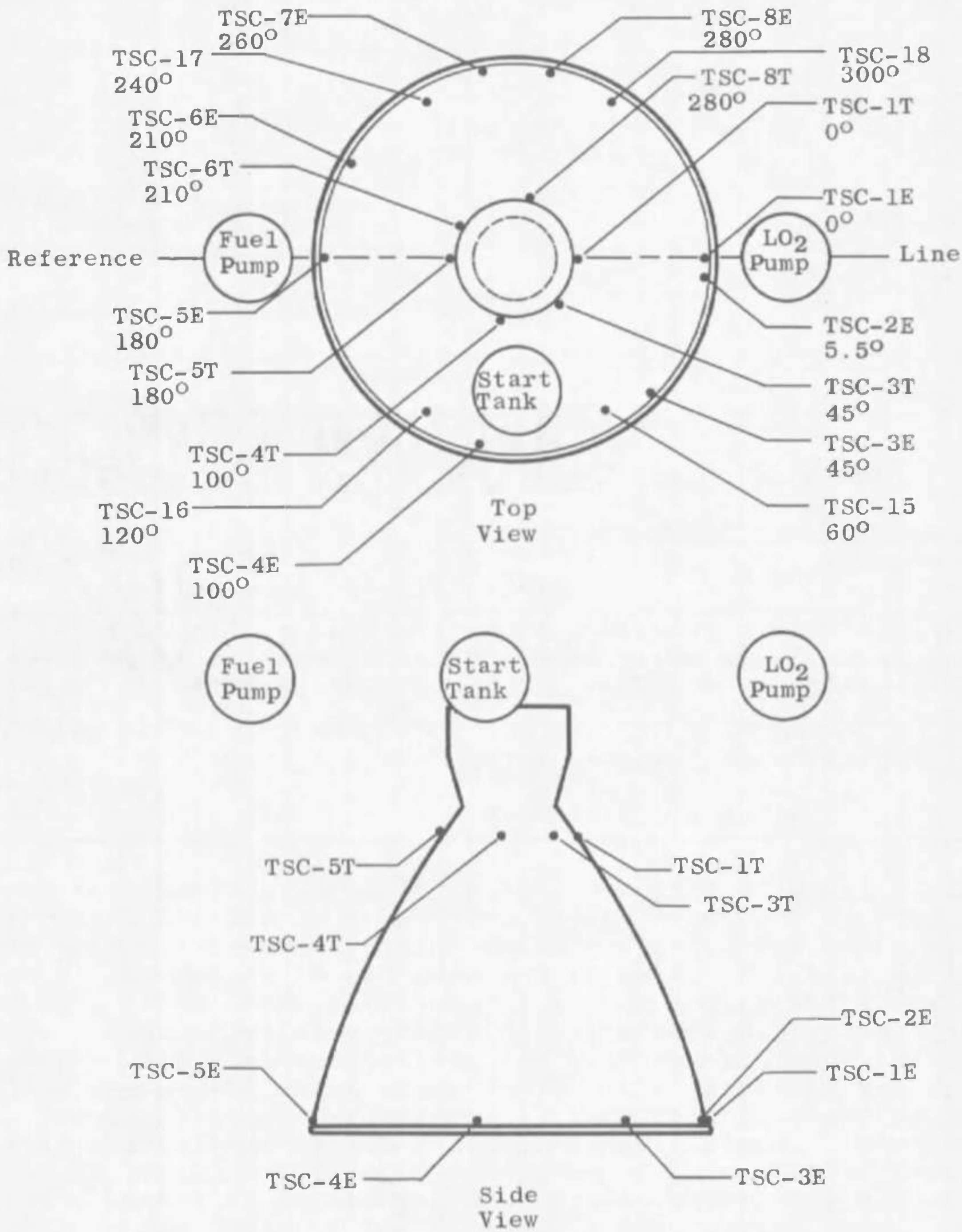
a. Engine Pressure Tap Locations

Fig. 13 Instrumentation Locations of the J-2 Engine



b. Engine Temperature, Flow, and Speed Instrumentation Locations

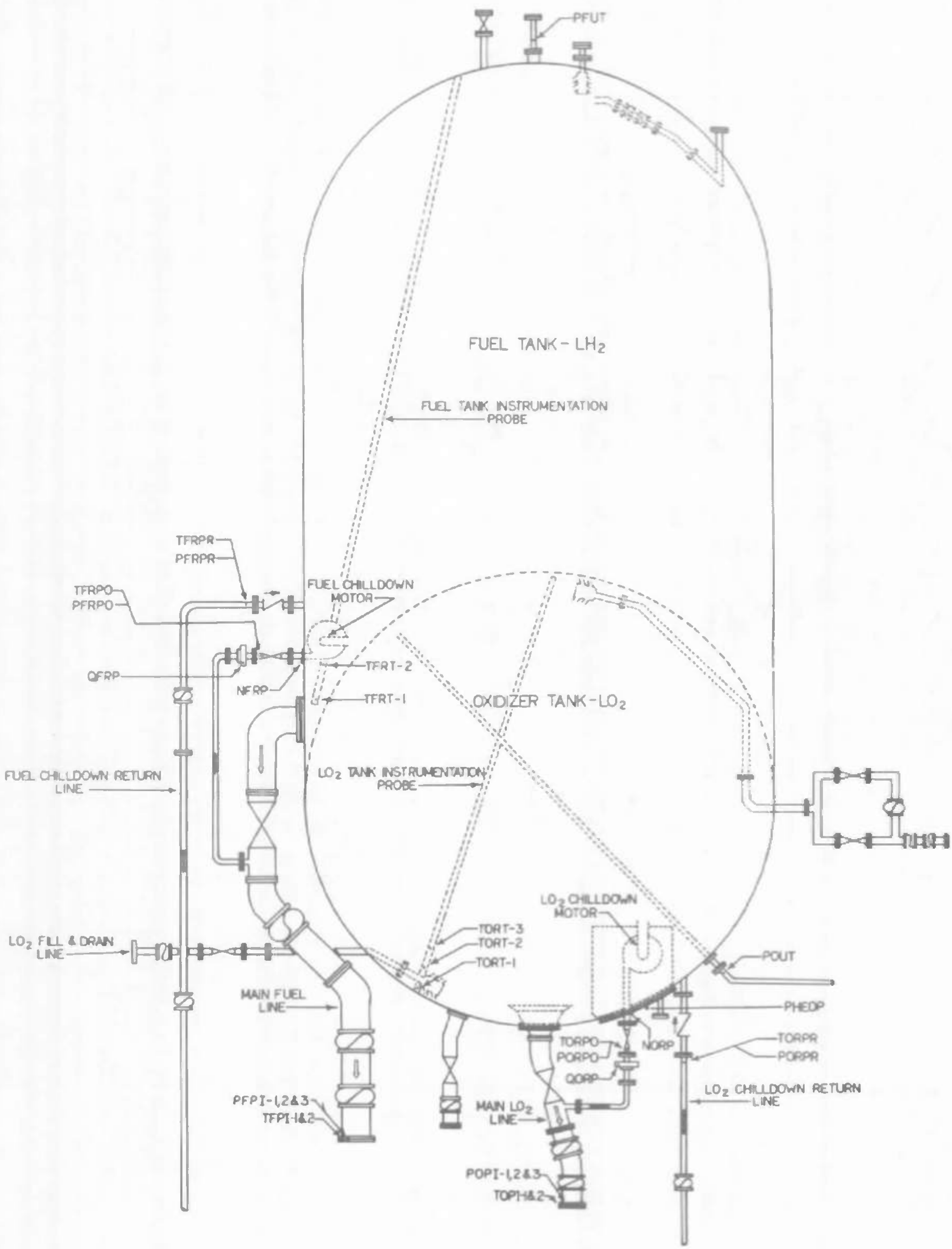
Fig. 13 Continued



NOTE: All angular locations are measured clockwise from O/F pump centerline.

c. Thrust Chamber

Fig. 13 Continued



d. S-IVB Stage Instrumentation

Fig. 13 Concluded

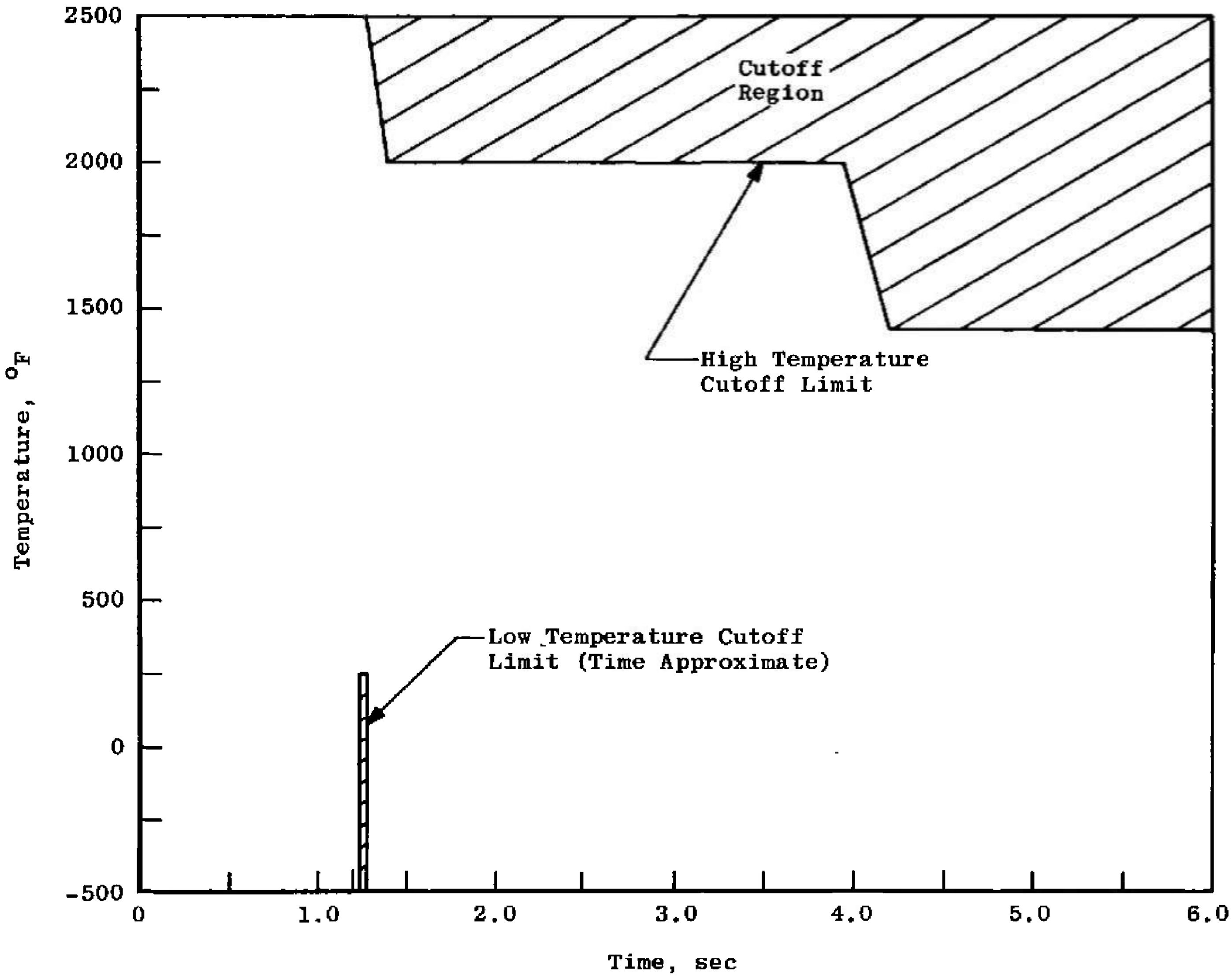


Fig. 14 Maximum and Minimum Allowable Limits for GGOT

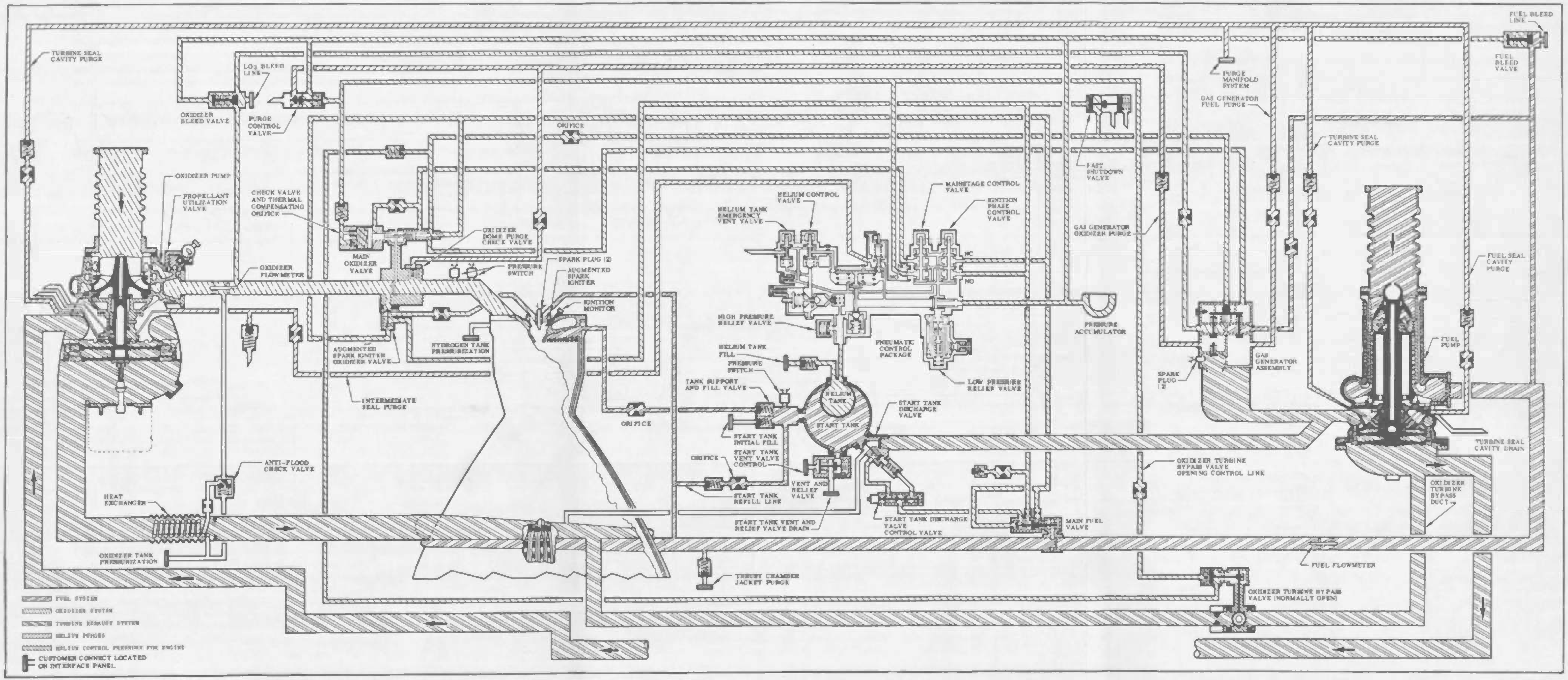


Fig. 15 Mechanical Schematic of the J-2 Engine

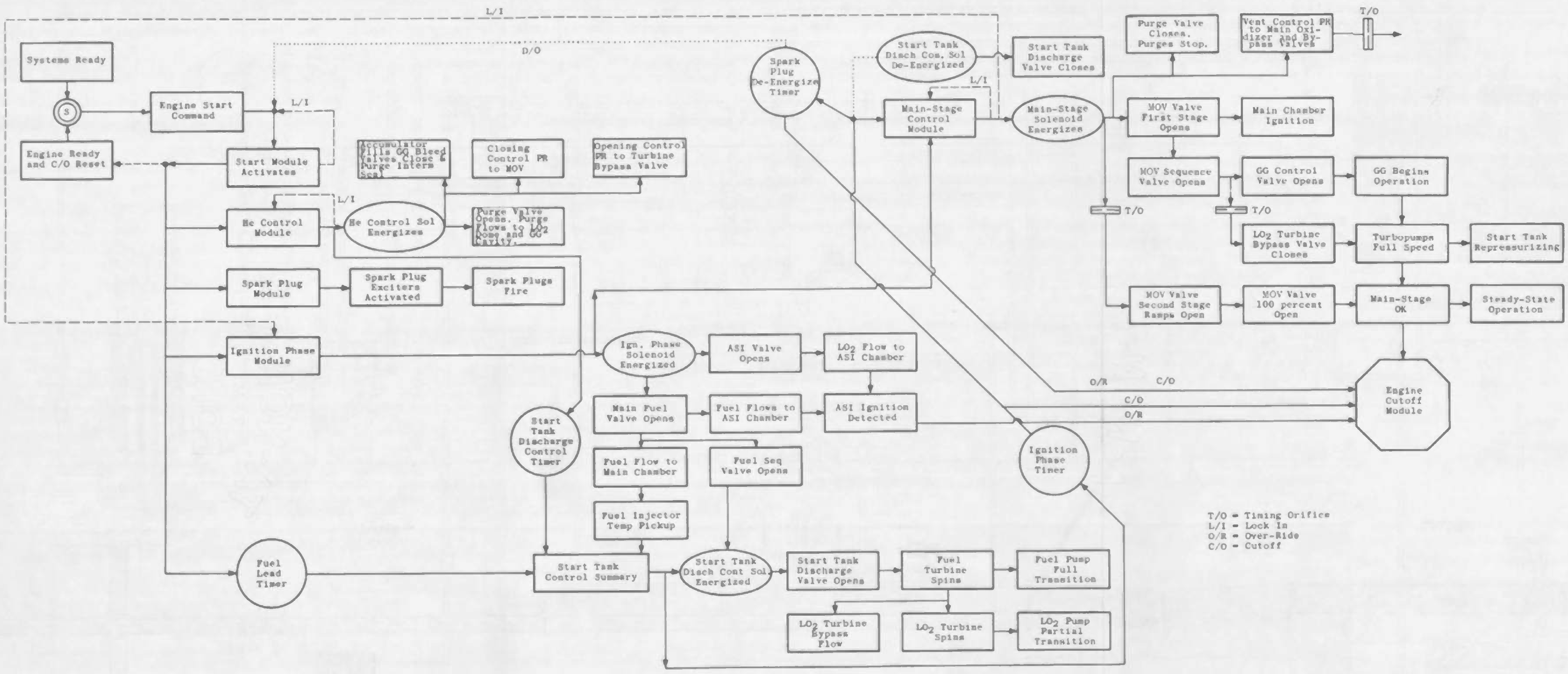


Fig. 16 Logic Schematic of the J-2 Engine Start

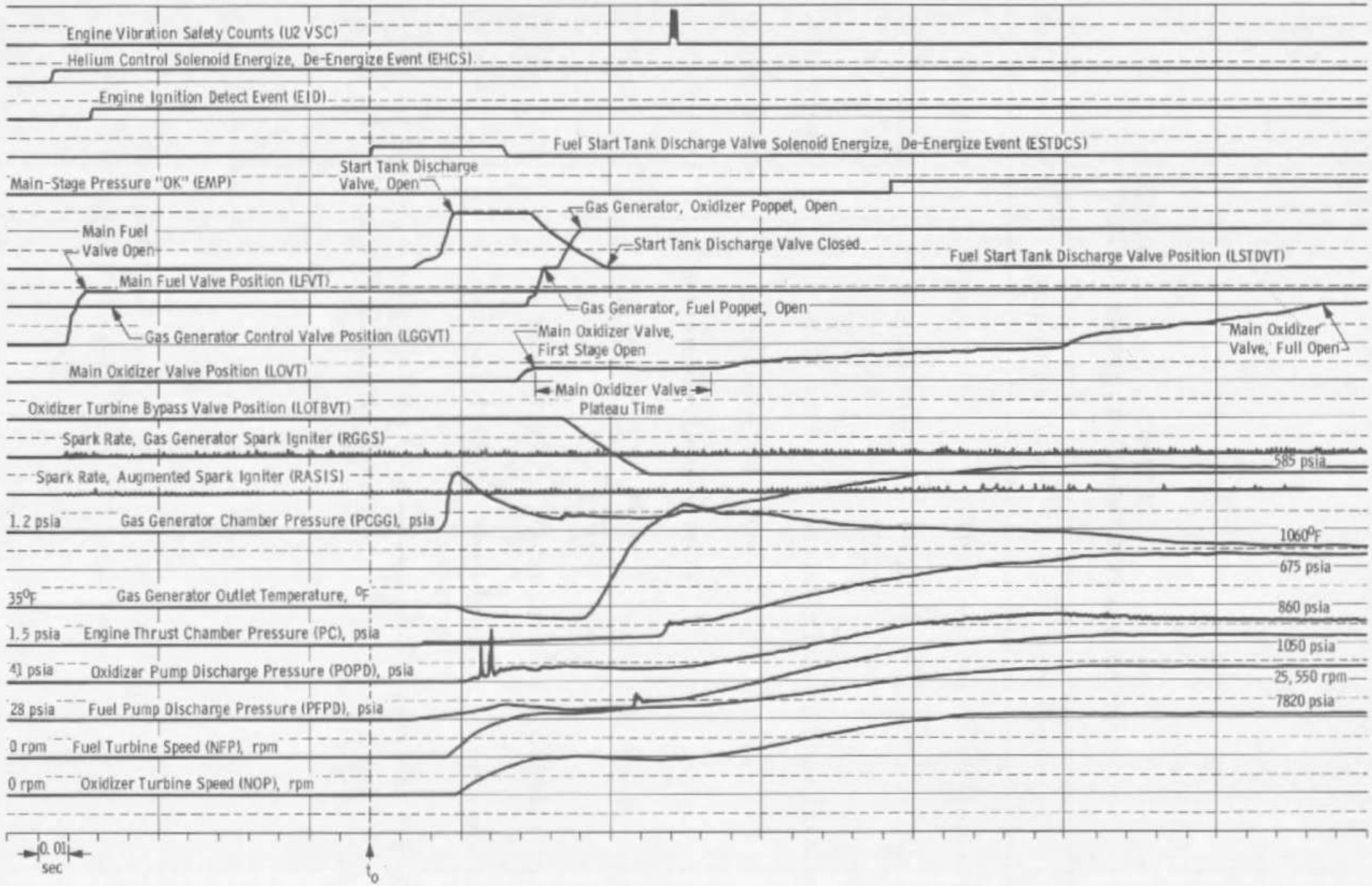
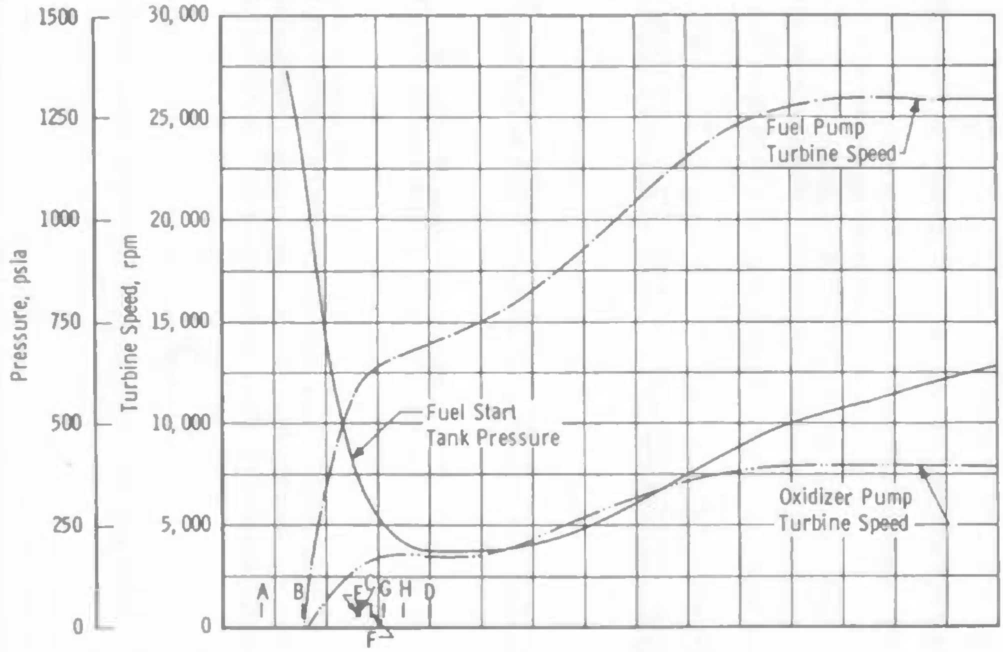
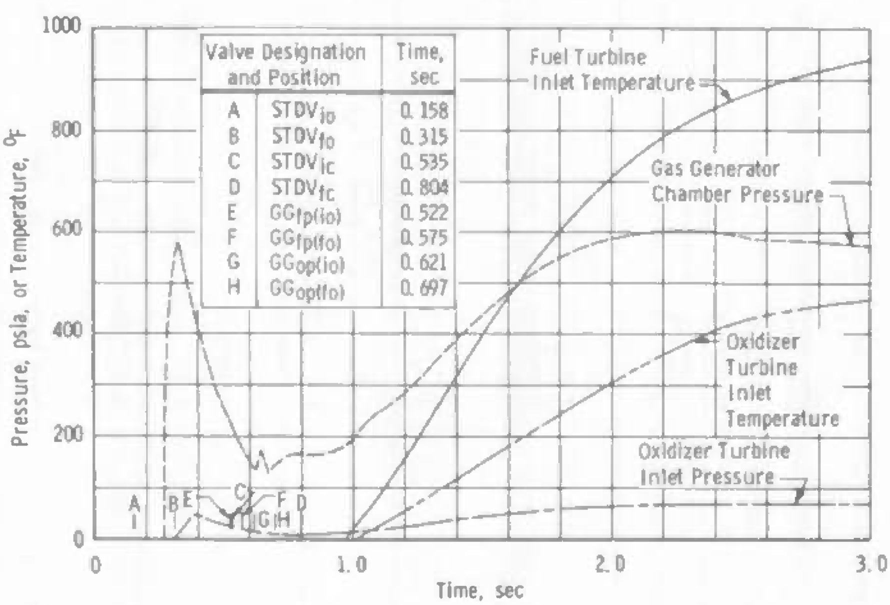


Fig. 17 Selected Engine Parameters for a Typical J-2 Transient



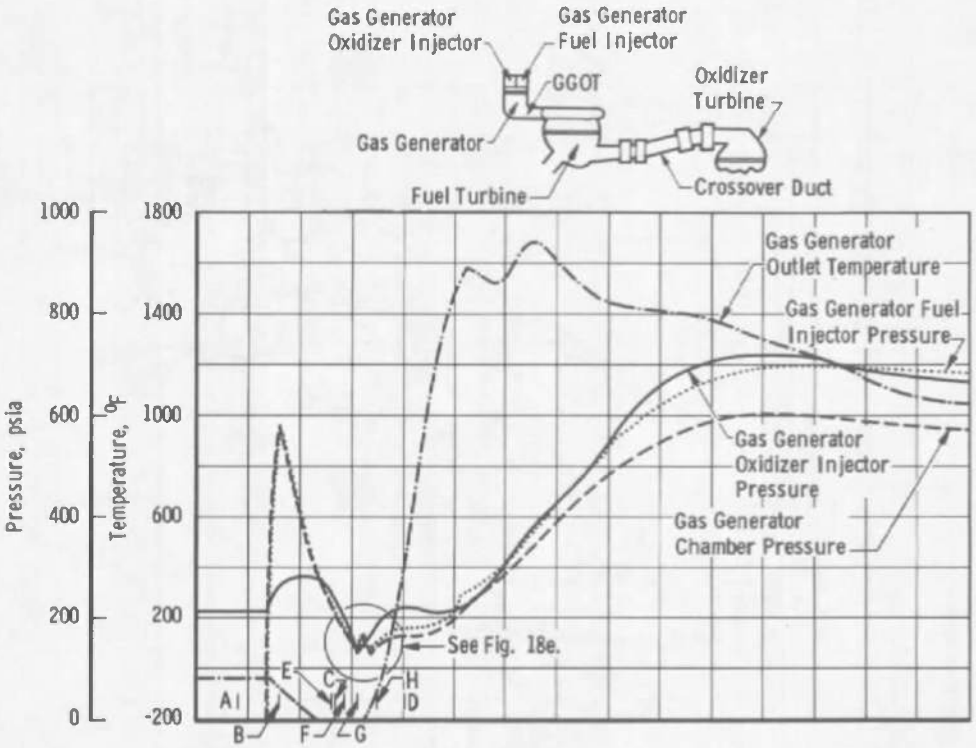
a. Fuel and Oxidizer Pump Turbine Spinup



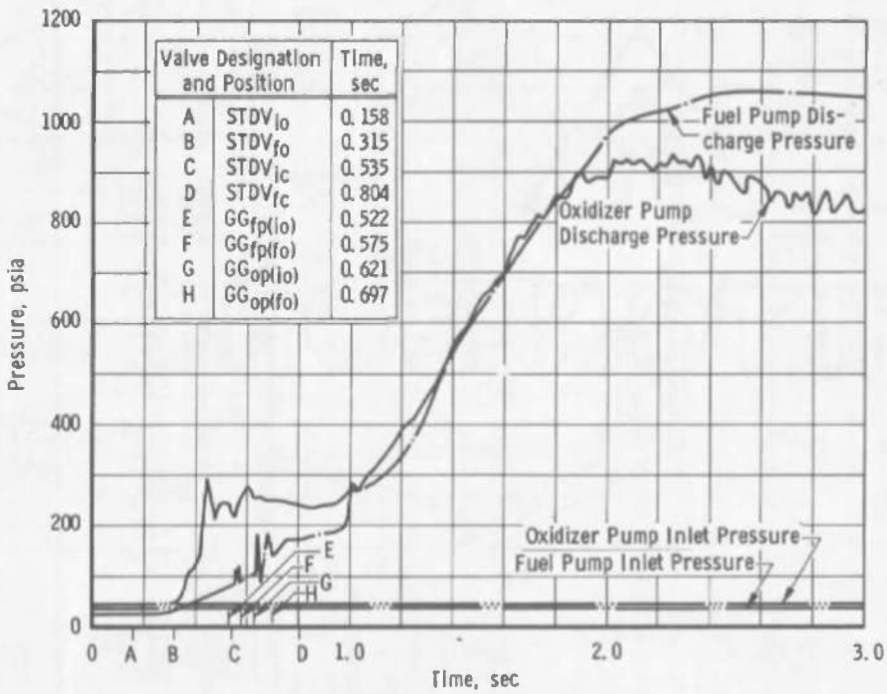
Valve Designation and Position		Time, sec
A	$STDV_{io}$	0.158
B	$STDV_{fo}$	0.315
C	$STDV_{ic}$	0.535
D	$STDV_{fc}$	0.804
E	$GG_{fp(io)}$	0.522
F	$GG_{fp(fo)}$	0.575
G	$GG_{op(io)}$	0.621
H	$GG_{op(fo)}$	0.697

b. Fuel and Oxidizer Turbine Inlet Pressures and Temperatures

Fig. 18 Start Transient Comparison of Selected Engine Parameters, Firing 10B

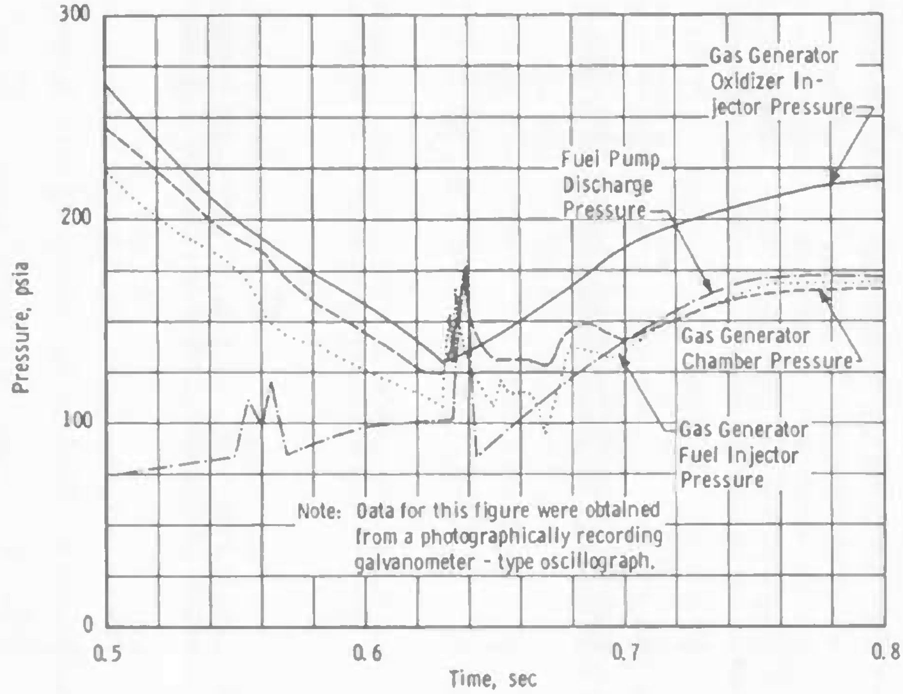


c. Gas Generator Pressures and Temperatures

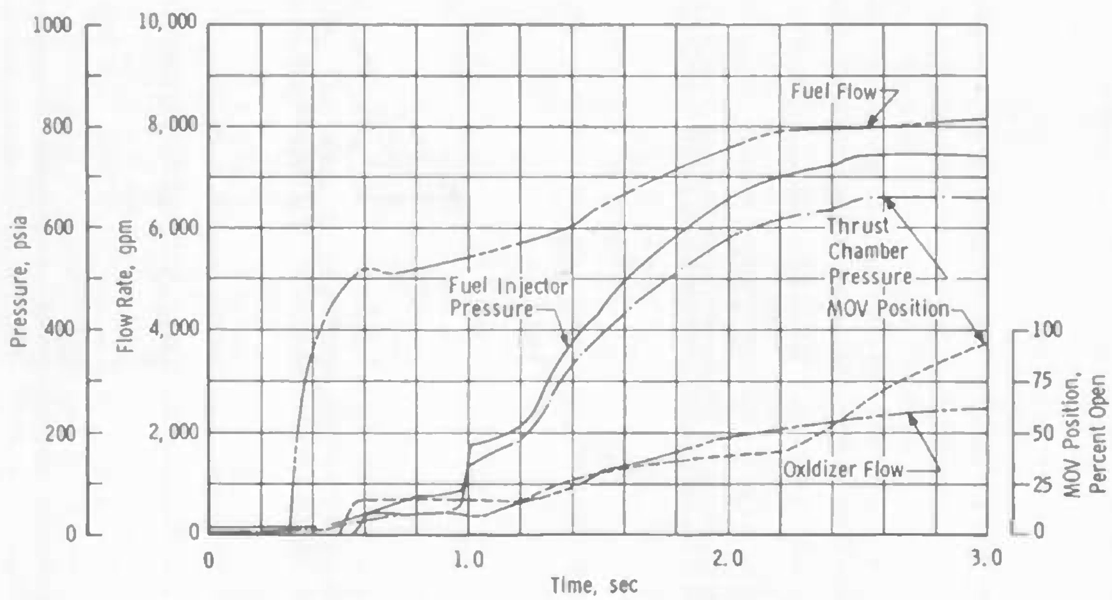


d. Fuel and Oxidizer Pump Inlet and Discharge Pressures

Fig. 18 Continued

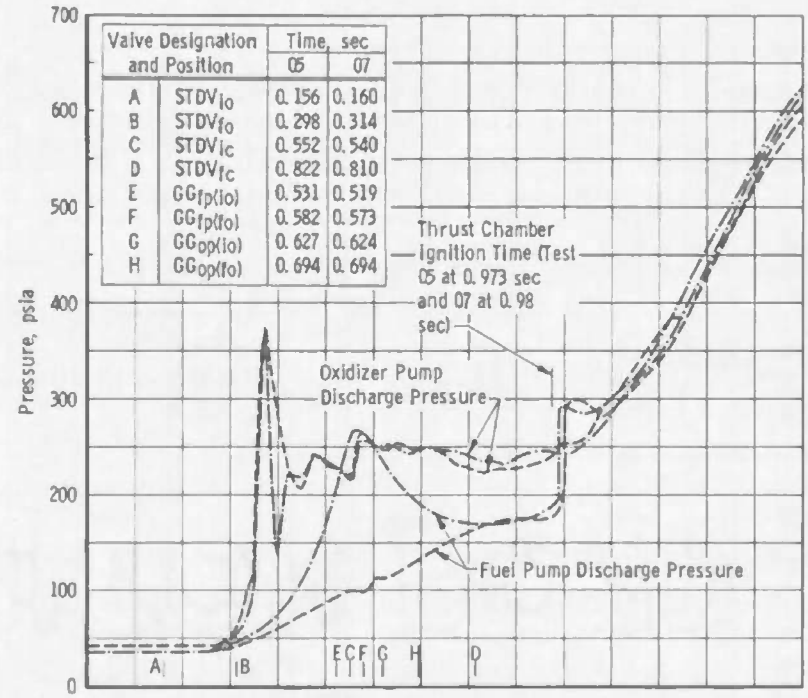


e. Gas Generator Pressures and Temperatures

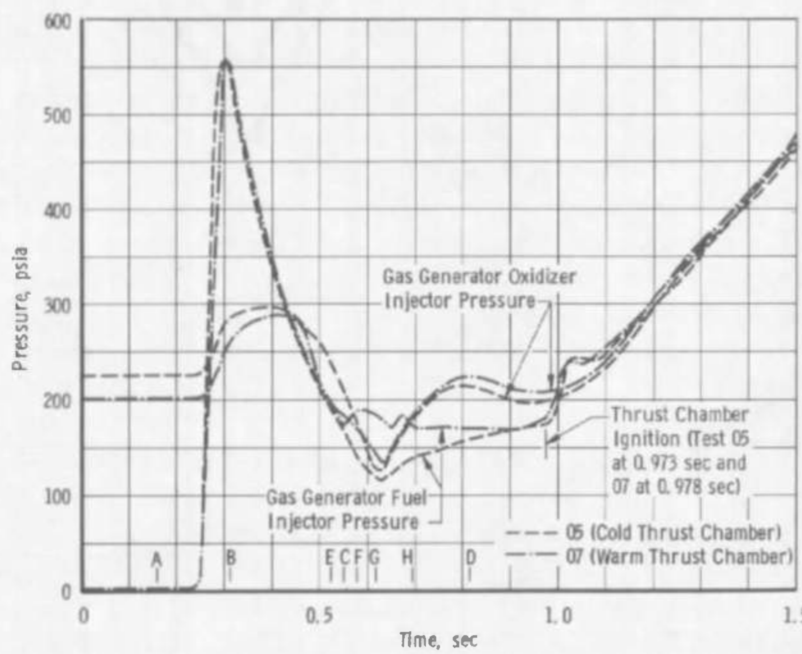


f. Thrust Chamber Pressures, Propellant Flows, and Main Oxidizer Valve Position

Fig. 18 Concluded

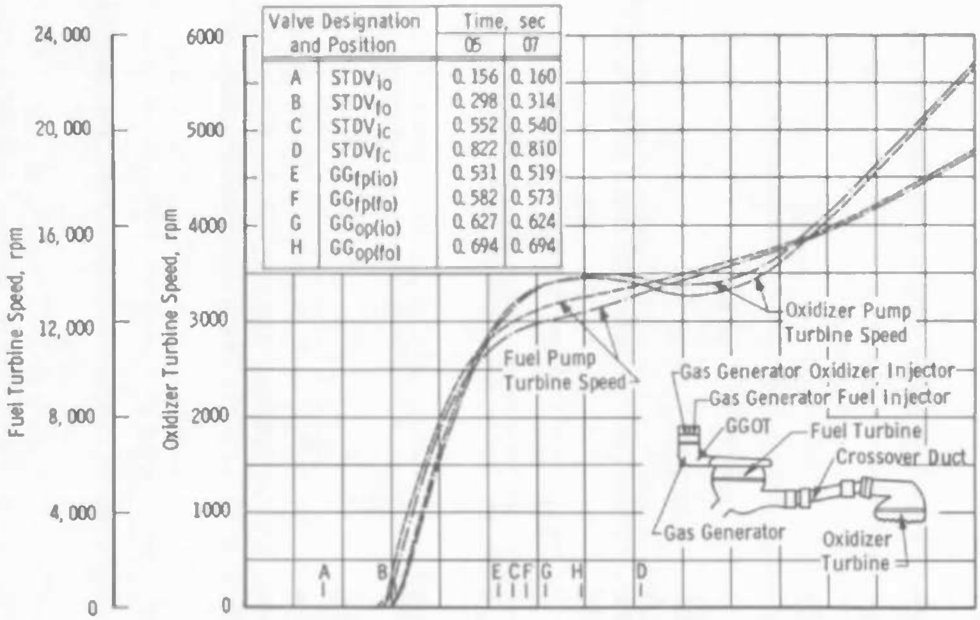


a. Fuel and Oxidizer Pump Discharge Pressures

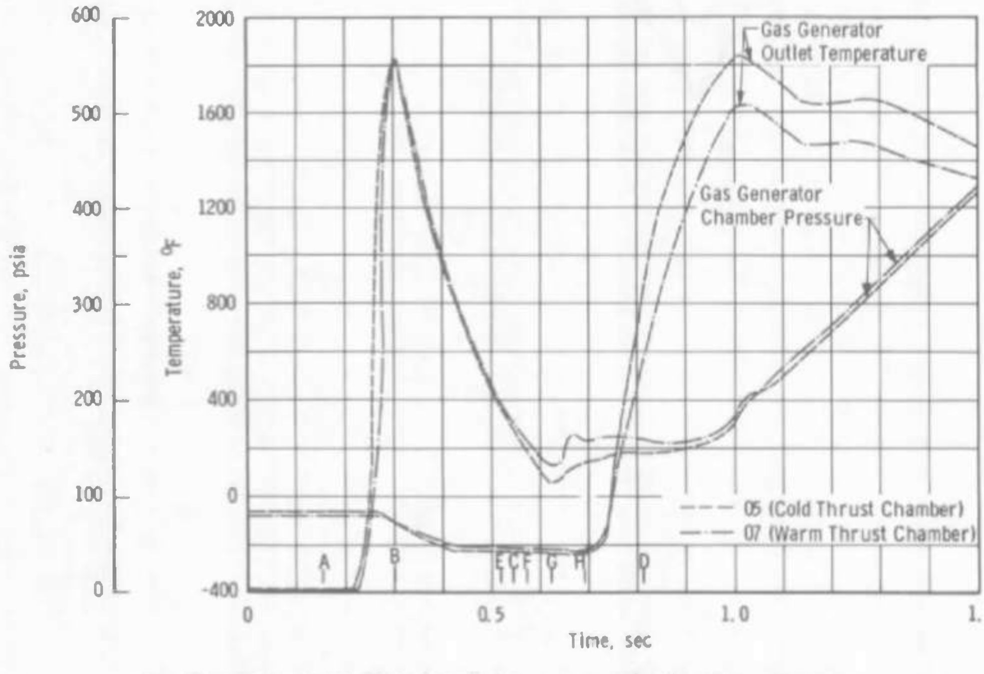


b. Gas Generator Fuel and Oxidizer Injector Pressures

Fig. 19 Effects of Fuel System Resistance on ES Transient Operation

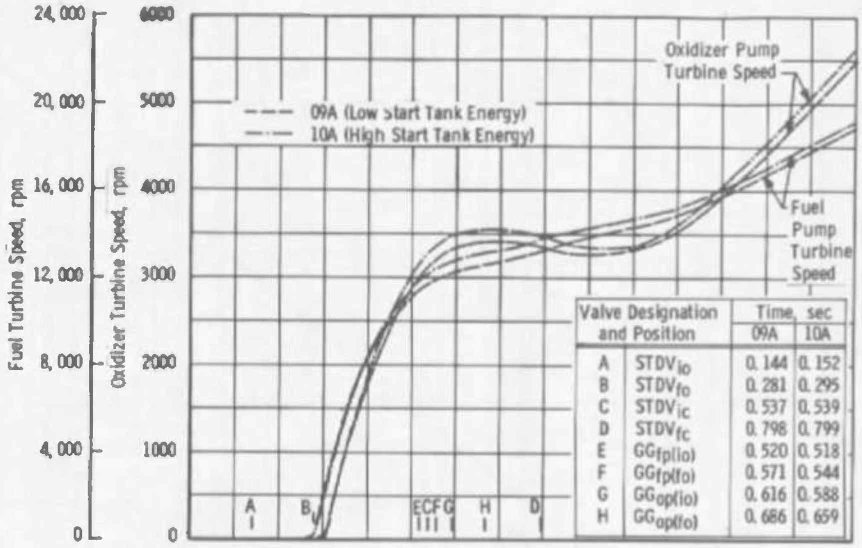


c. Fuel and Oxidizer Pump Turbine Speeds

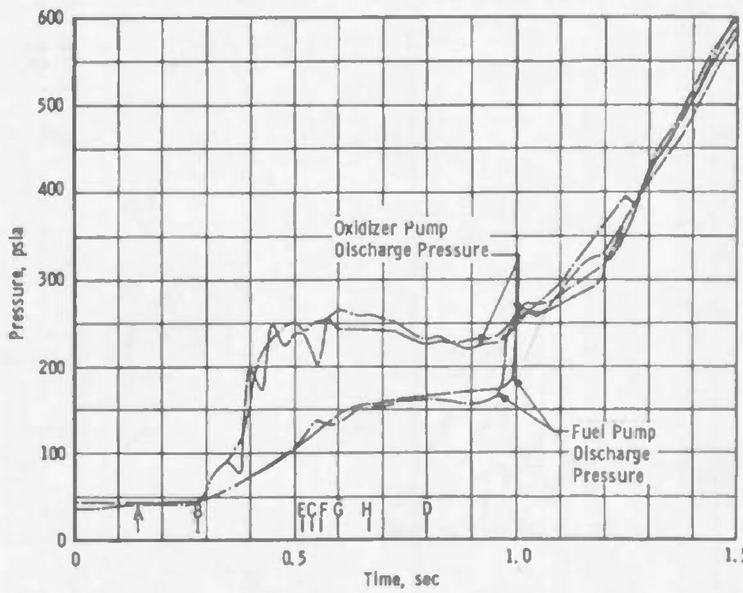


d. Gas Generator Chamber Pressure and Outlet Temperatures

Fig. 19 Concluded

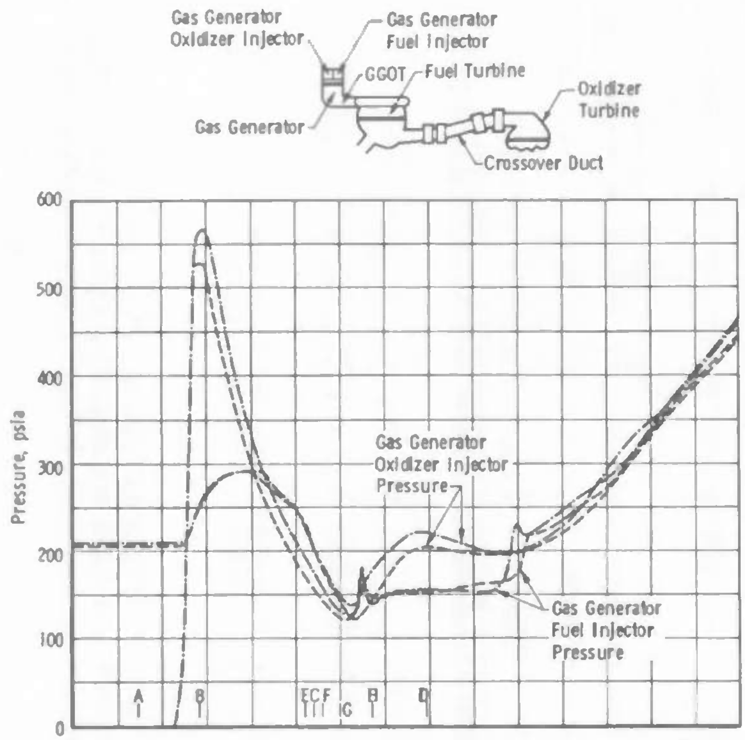


a. Fuel and Oxidizer Pump Turbine Speeds

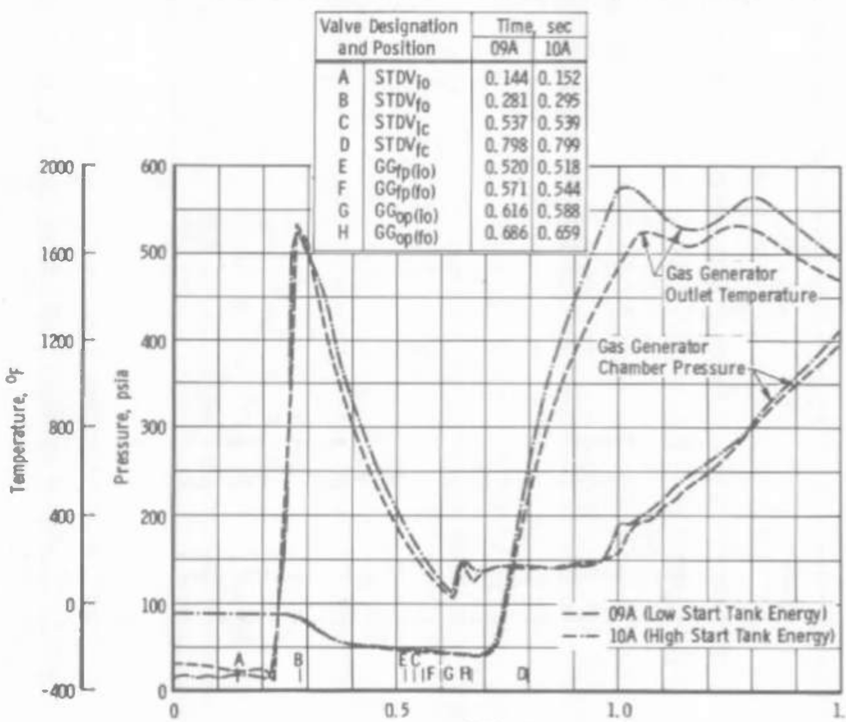


b. Fuel and Oxidizer Pump Discharge Pressure

Fig. 20 Effects of Fuel Start Tank Energy on ES Transient Operation



c. Gas Generator Fuel and Oxidizer Ignition Pressures



Valve Designation and Position		Time, sec 09A	10A
A	$STDV_{io}$	0.144	0.152
B	$STDV_{fo}$	0.281	0.295
C	$STDV_{ic}$	0.537	0.539
D	$STDV_{fc}$	0.798	0.799
E	$GG_{fp(io)}$	0.520	0.518
F	$GG_{fp(fo)}$	0.571	0.544
G	$GG_{op(io)}$	0.616	0.588
H	$GG_{op(fo)}$	0.686	0.659

d. Gas Generator Outlet Temperature and Chamber Pressure

Fig. 20 Concluded

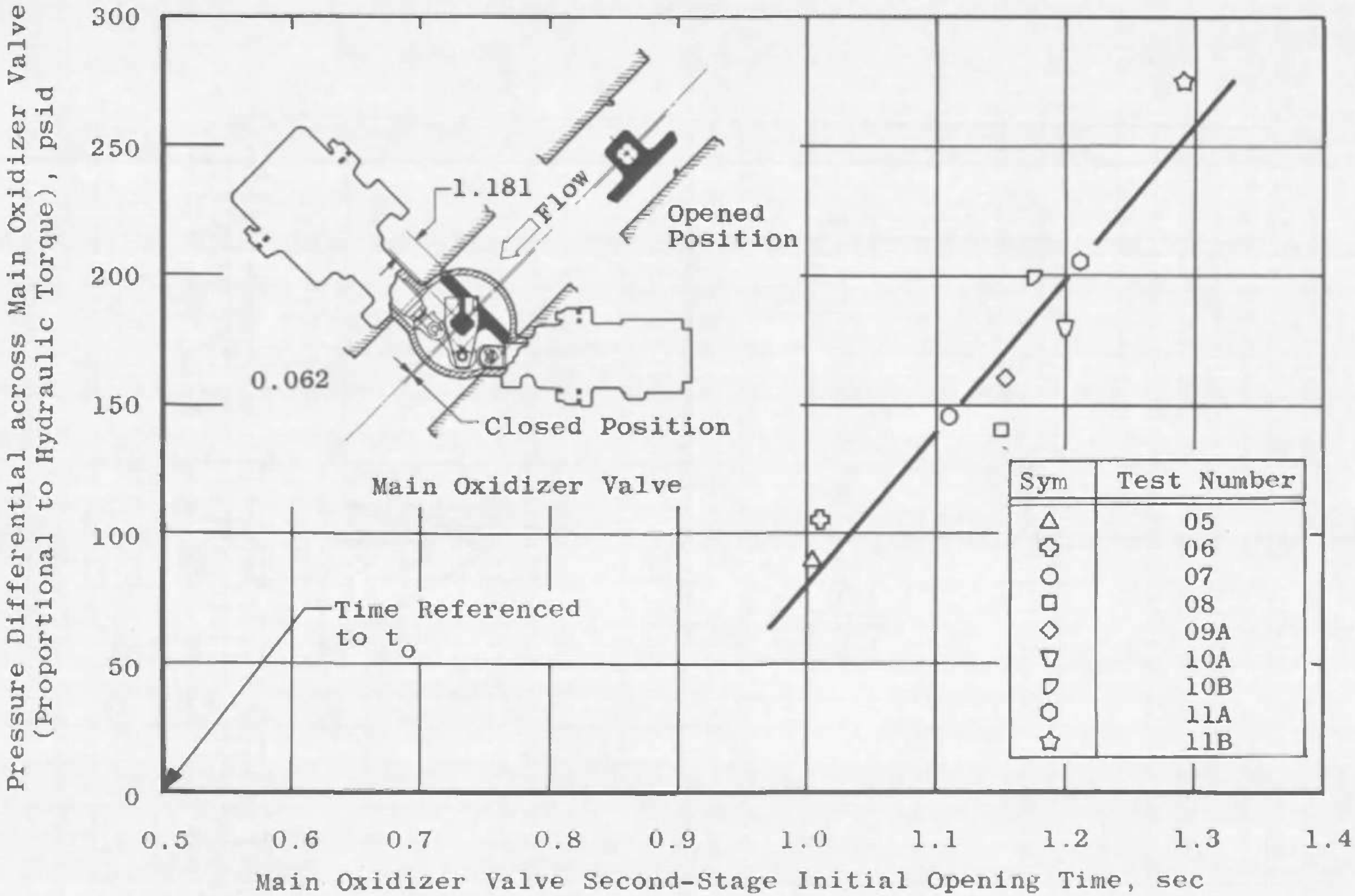


Fig. 21 Hydraulic Torque Effects on MOV Second-Stage Initial Opening Time

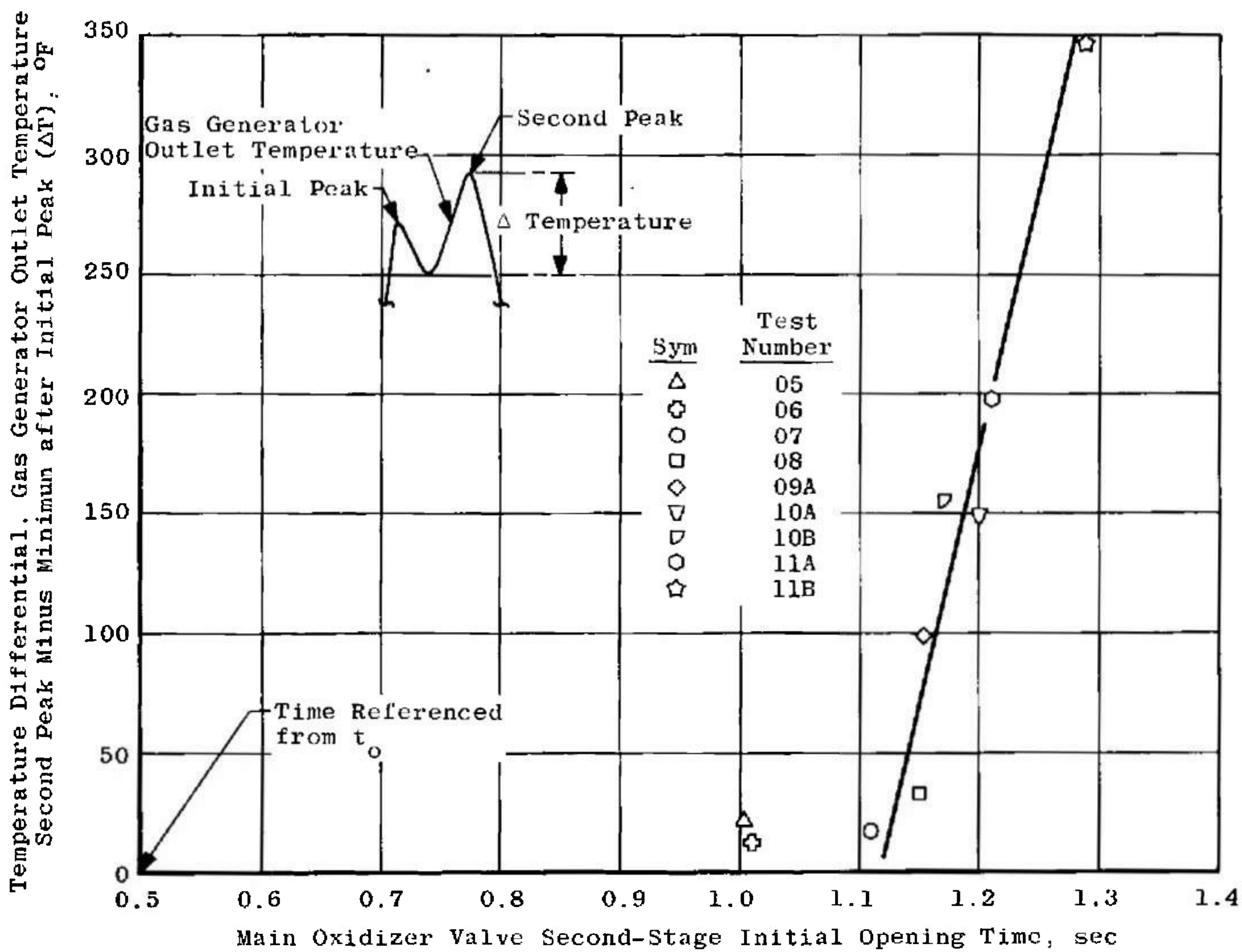


Fig. 22 Effects of MOV Second-Stage Opening Time on GGOT Second Peak

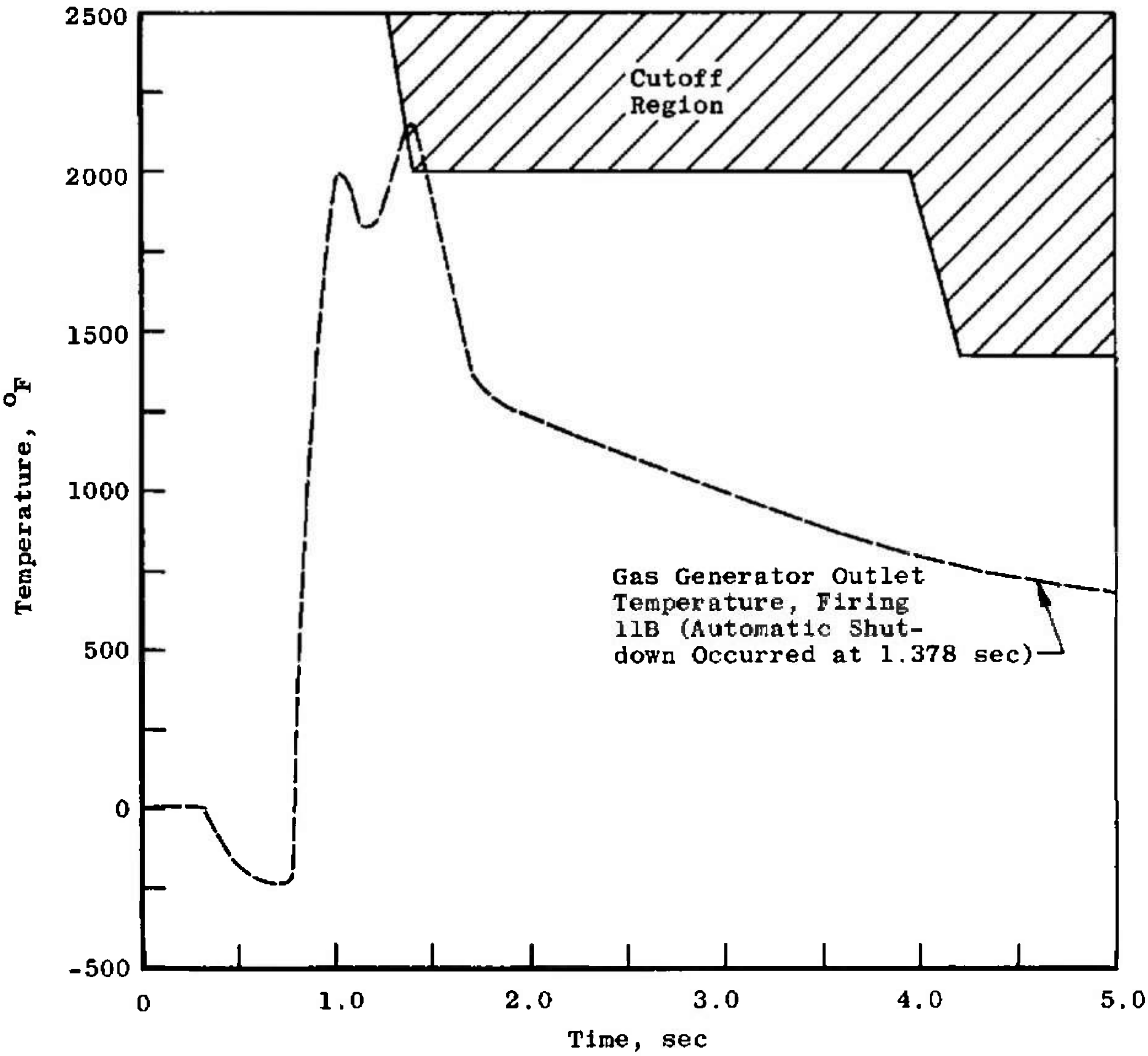


Fig. 23 GGOT, Start Transient, Firing 11B

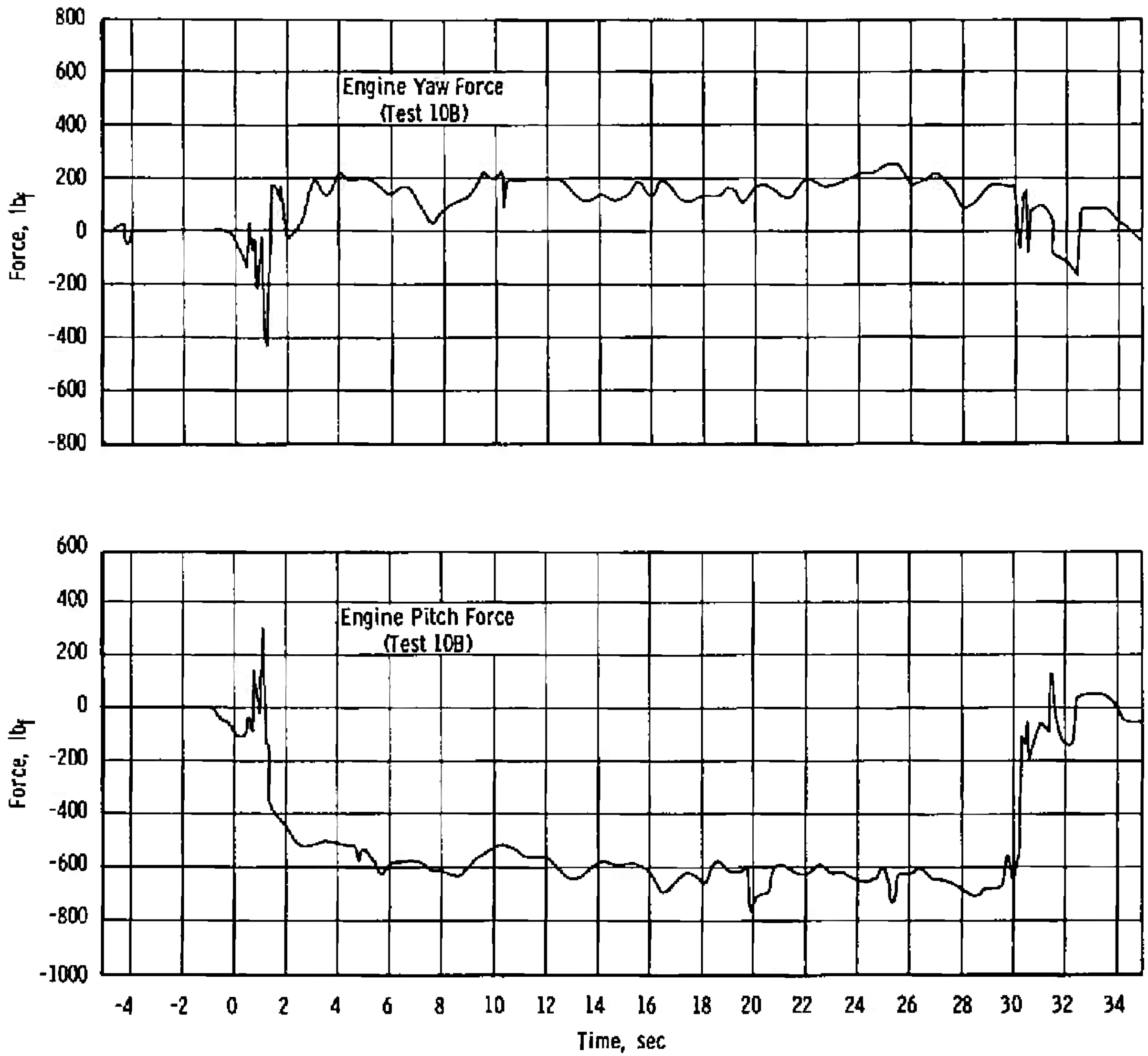


Fig. 24 Engine Side-Force Data

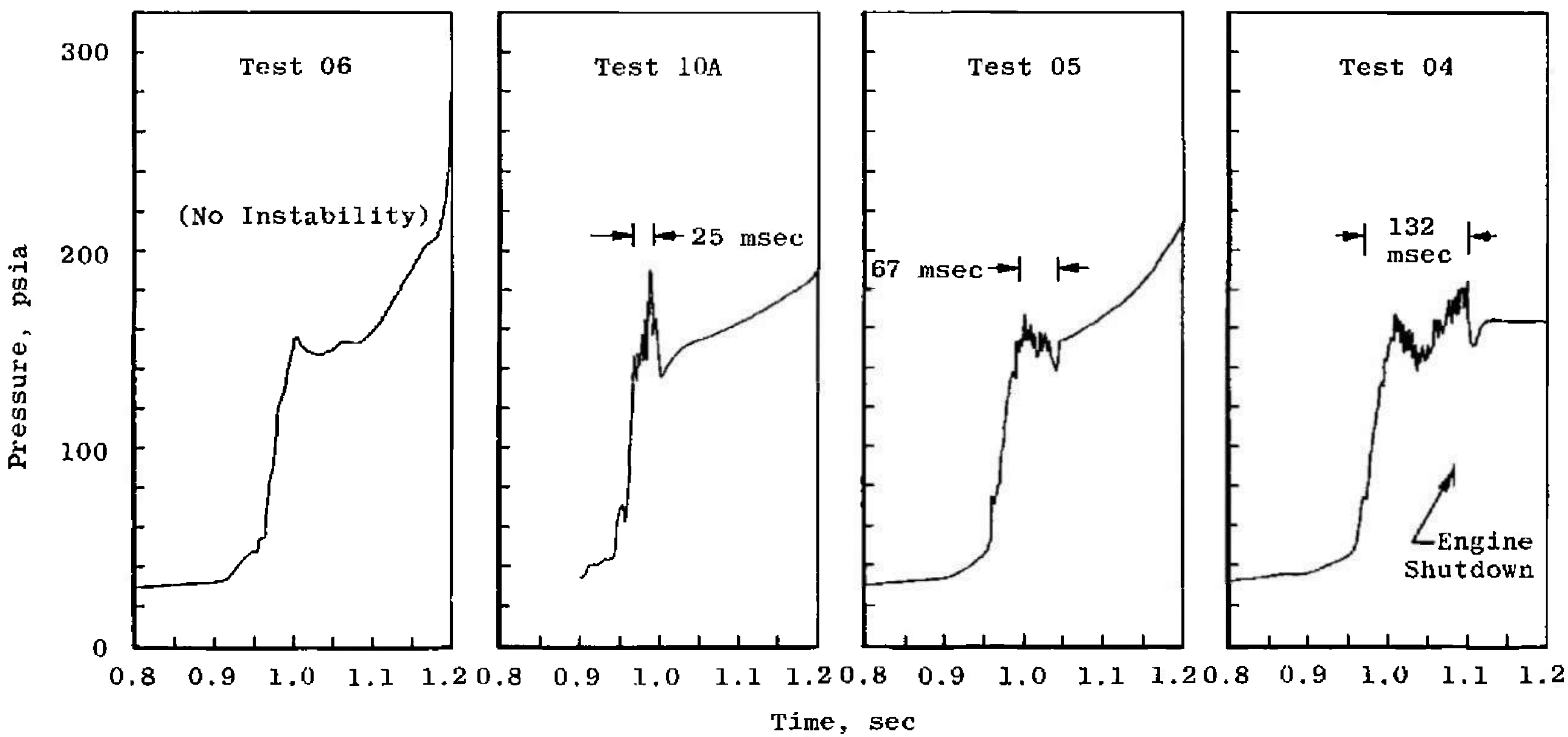


Fig. 25 Thrust Chamber Pressure Fluctuations during the ES Transient

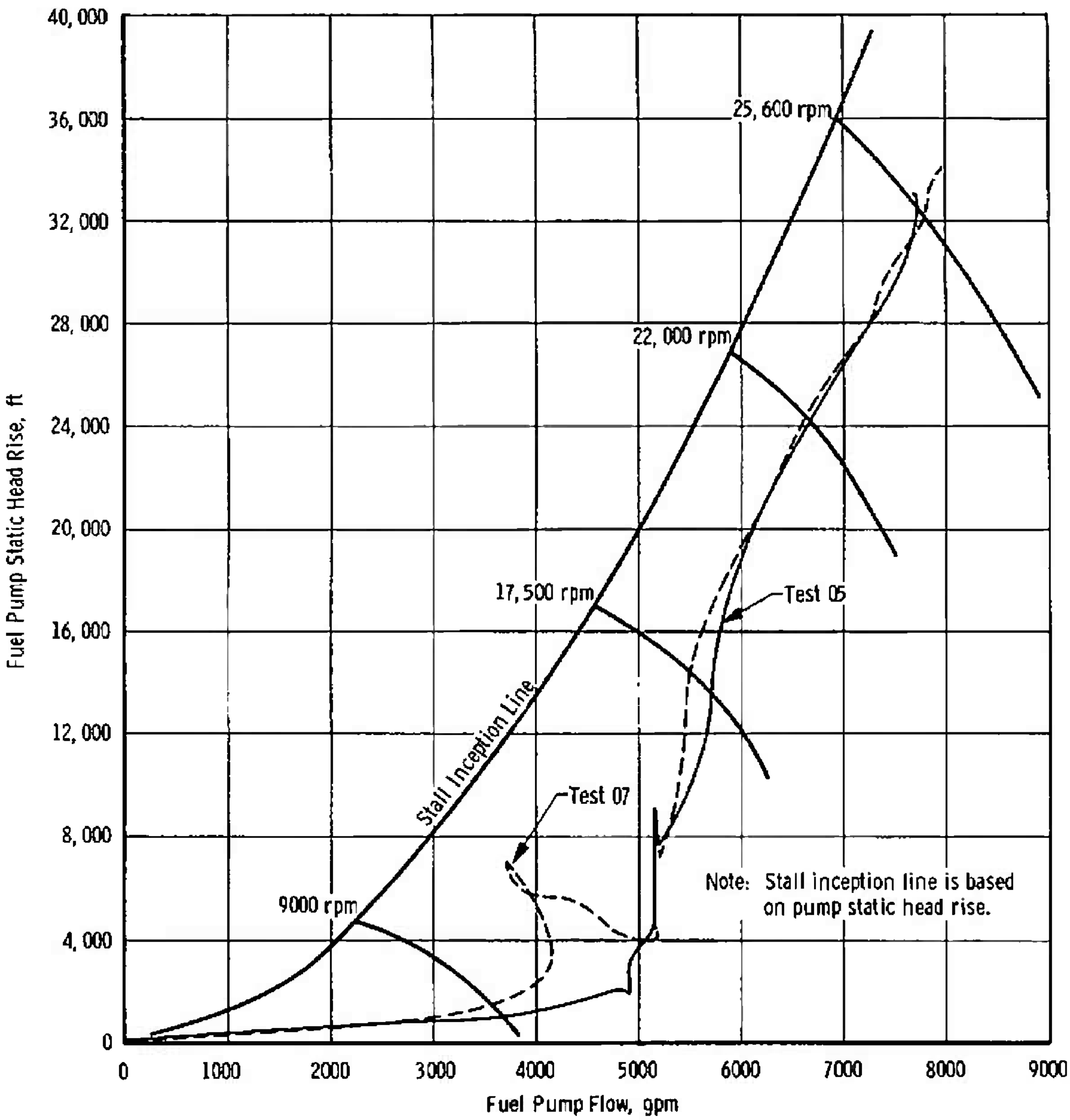
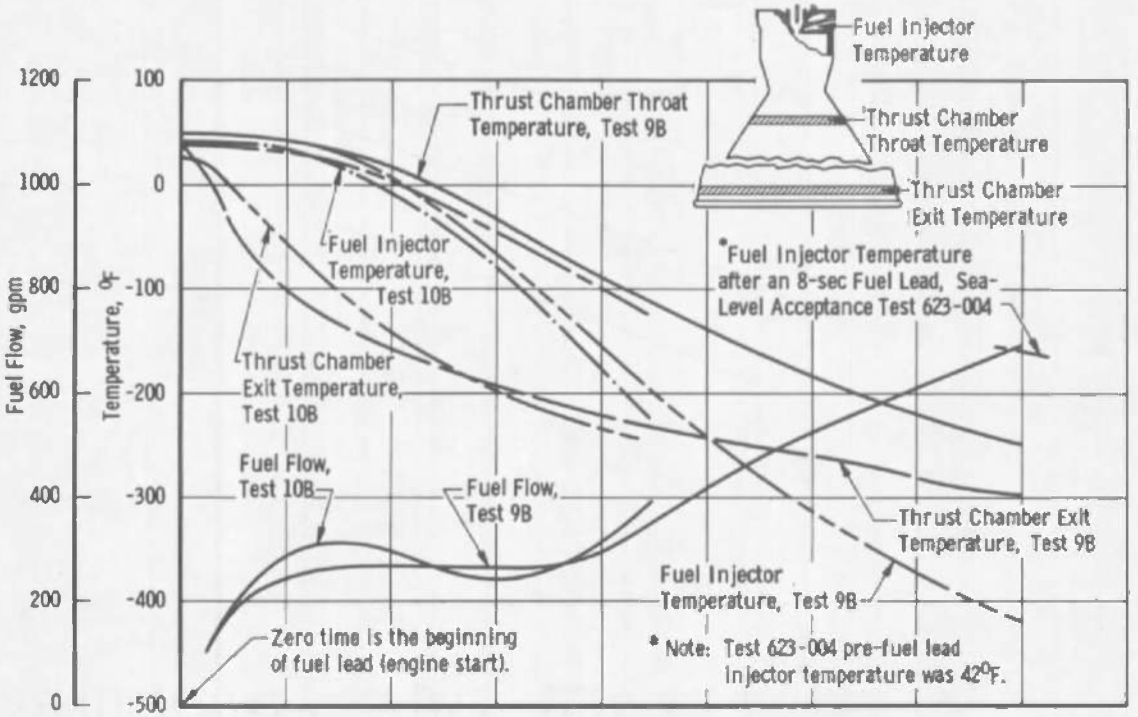
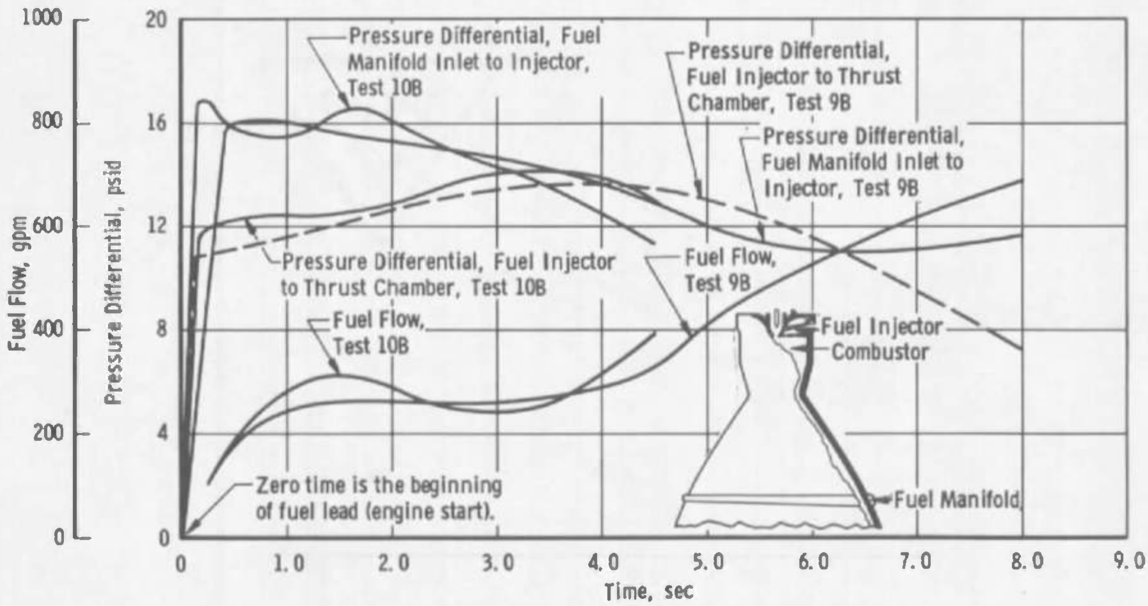


Fig. 26 Fuel Pump Start Transient Performance



a. Temperature and Fuel Flow Trends



b. Fuel System Pressure Differentials and Fuel Flow Trends

Fig. 27 Fuel Lead Effects

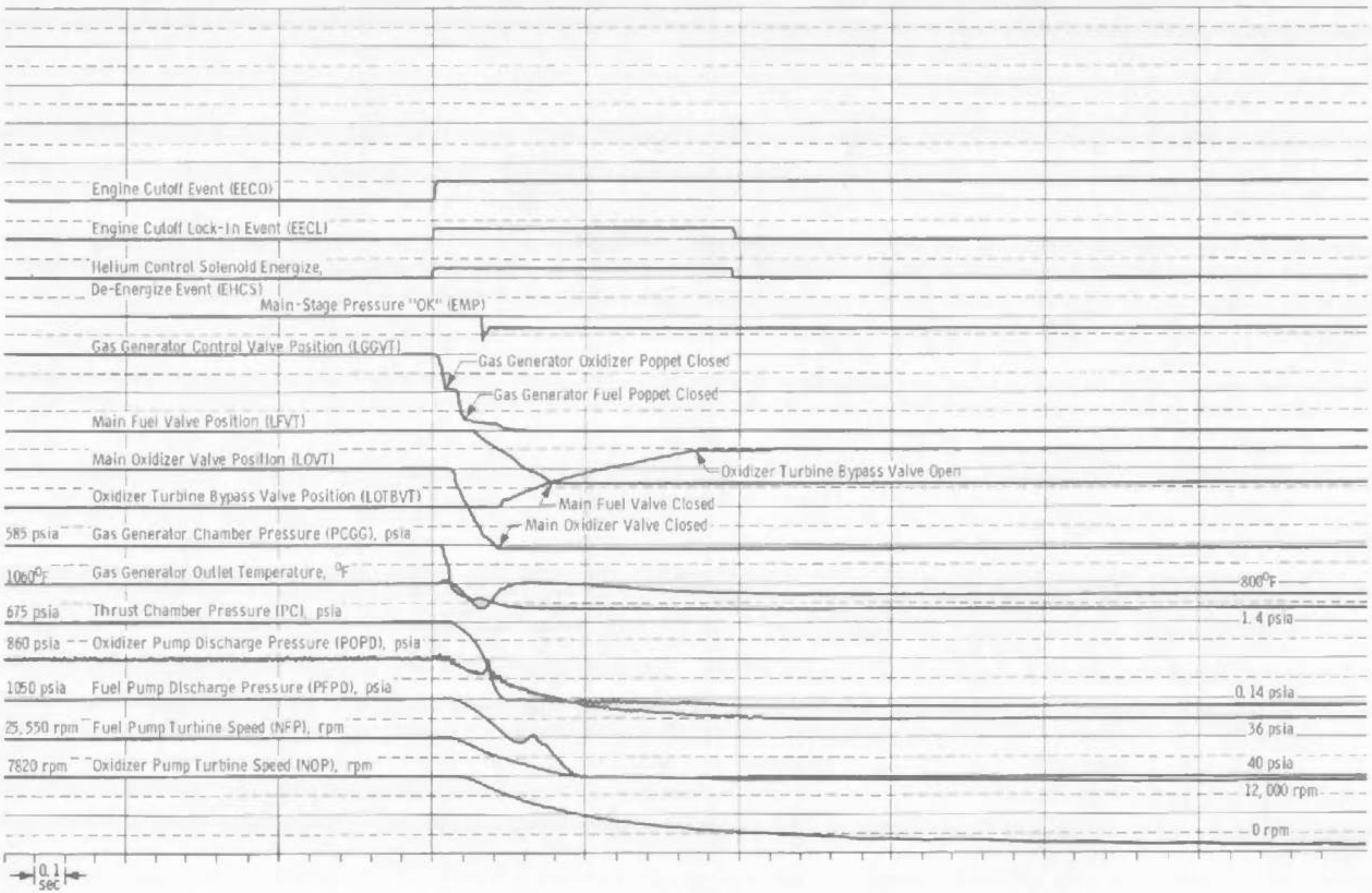
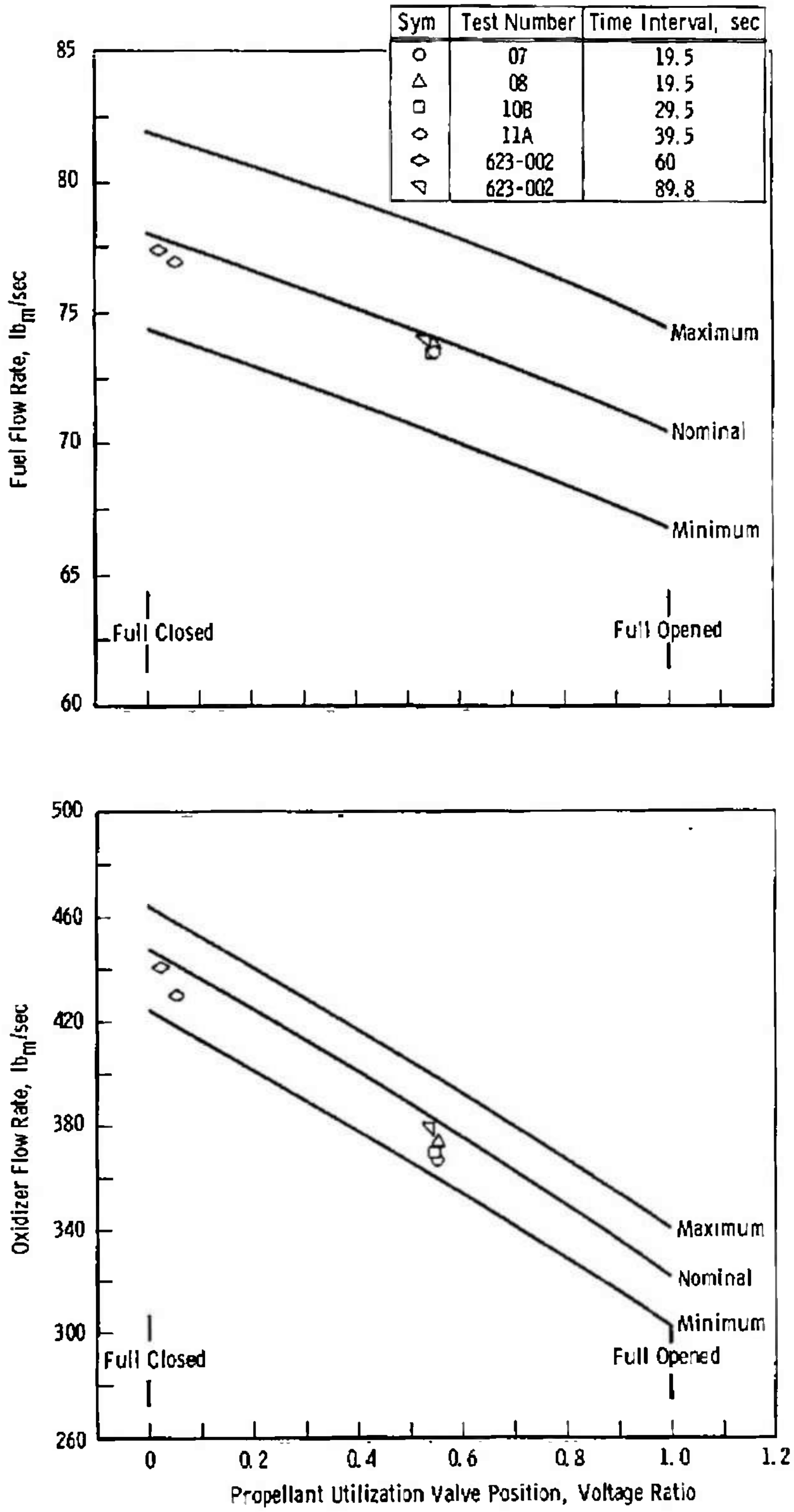
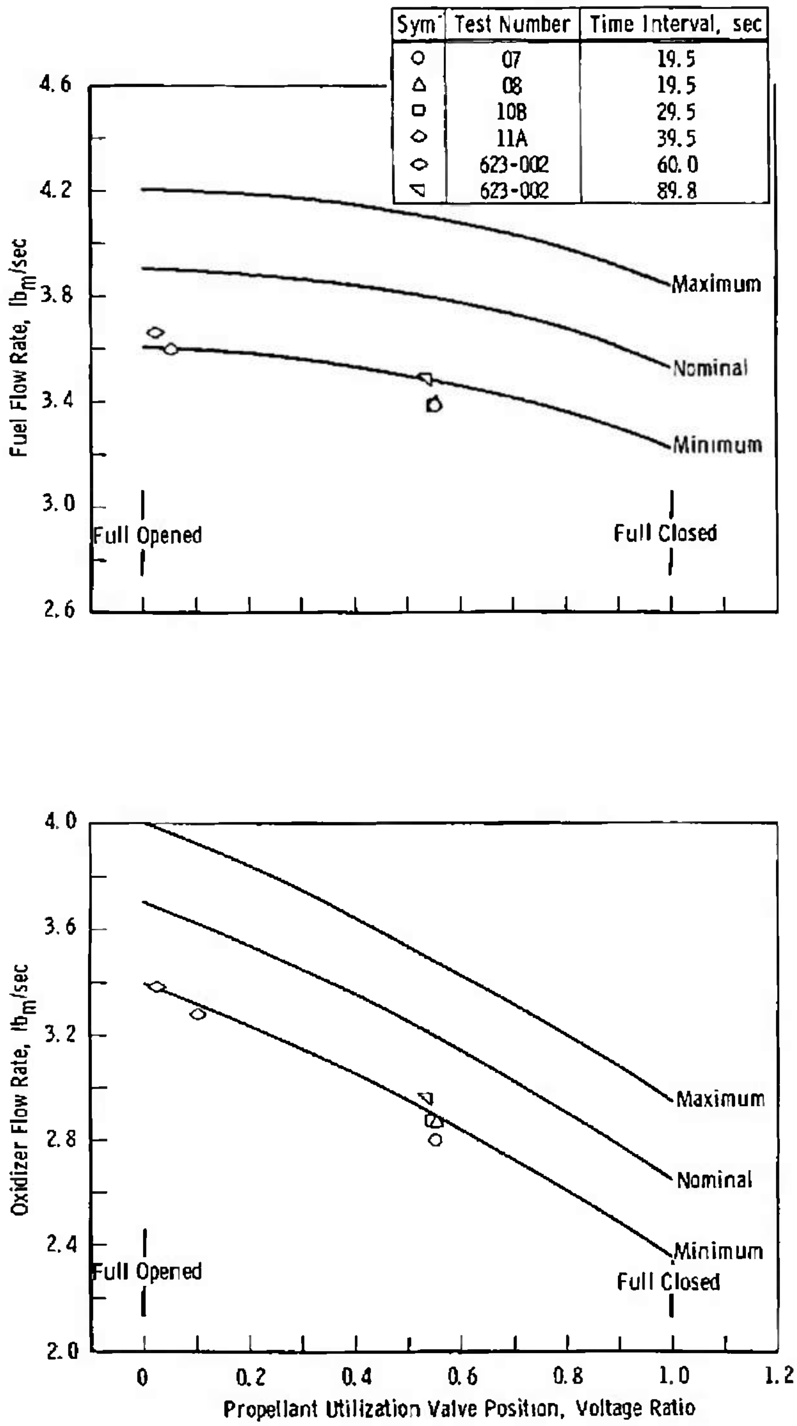


Fig. 28 Typical J-2 Engine Shutdown Transient



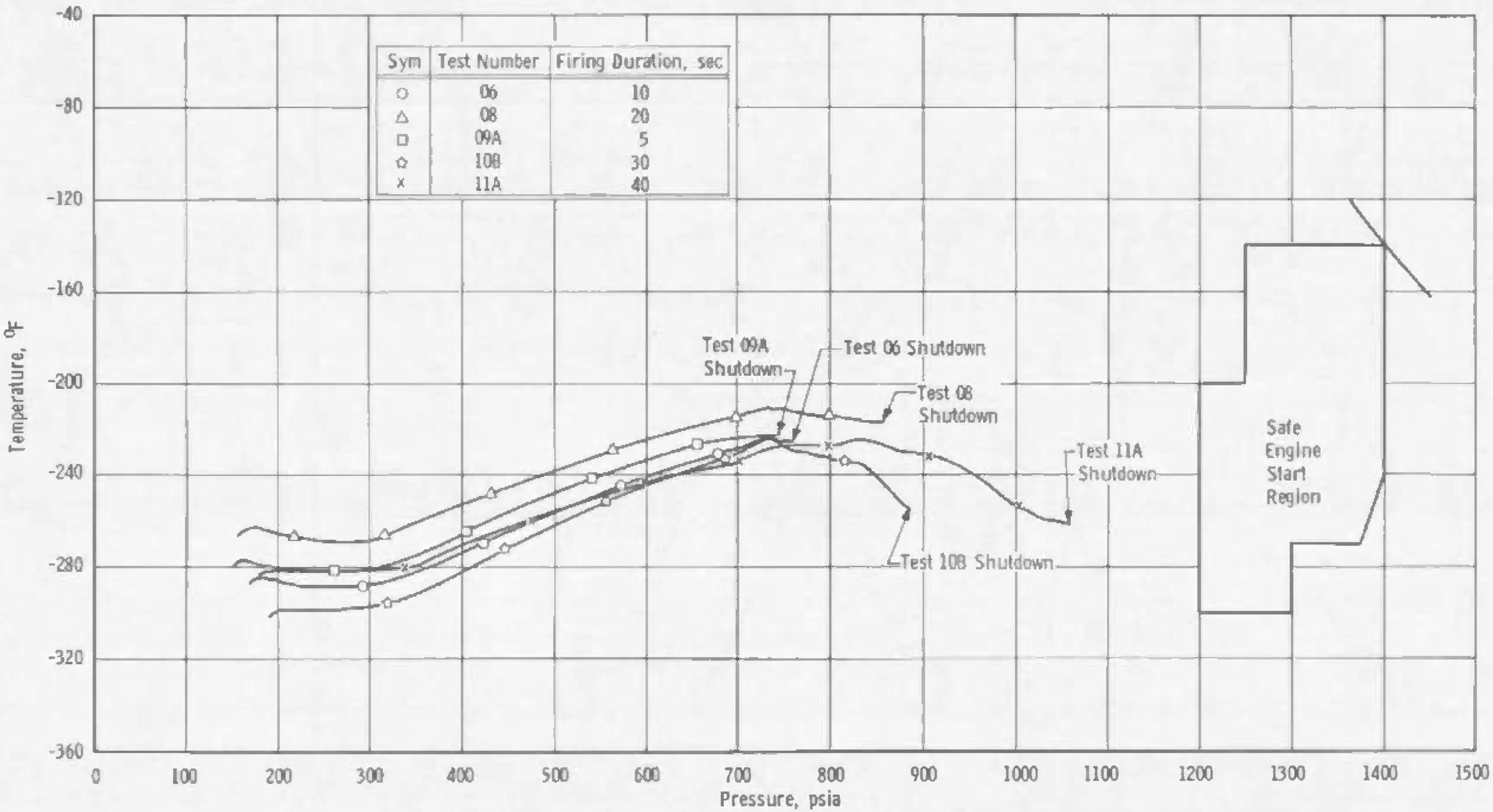
a. Thrust Chamber Flow Rate versus PU Valve Position

Fig. 29 Thrust Chamber and Gas Generator Propellant Flow Rates versus PU Valve Position



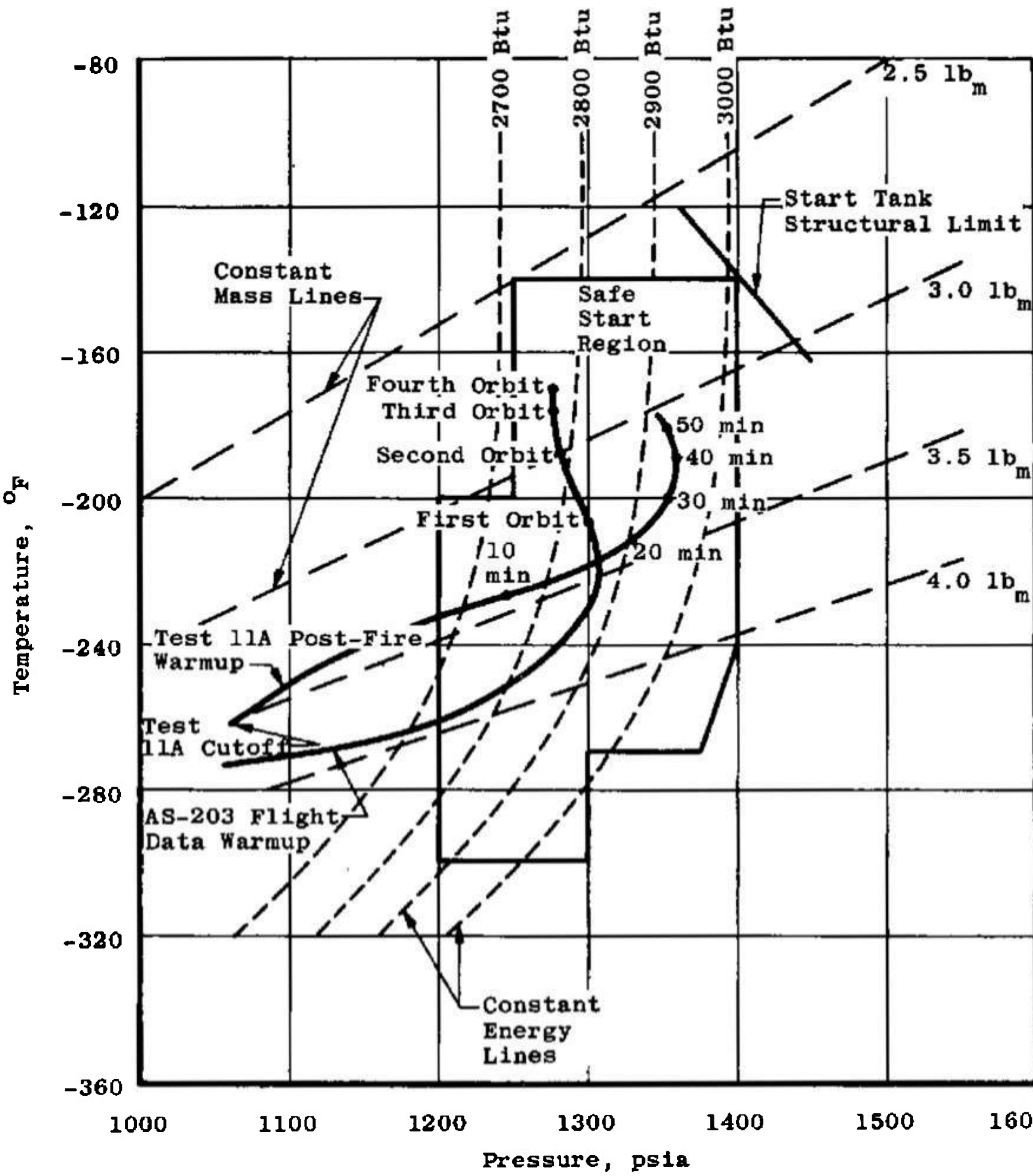
b. Gas Generator Flow Rate versus PU Valve Position

Fig. 29 Concluded



a. Start Tank Repressurization

Fig. 30 Fuel Start Tank Repressurization and Warmup Data



b. Start Tank Warmup Data

Fig. 30 Concluded

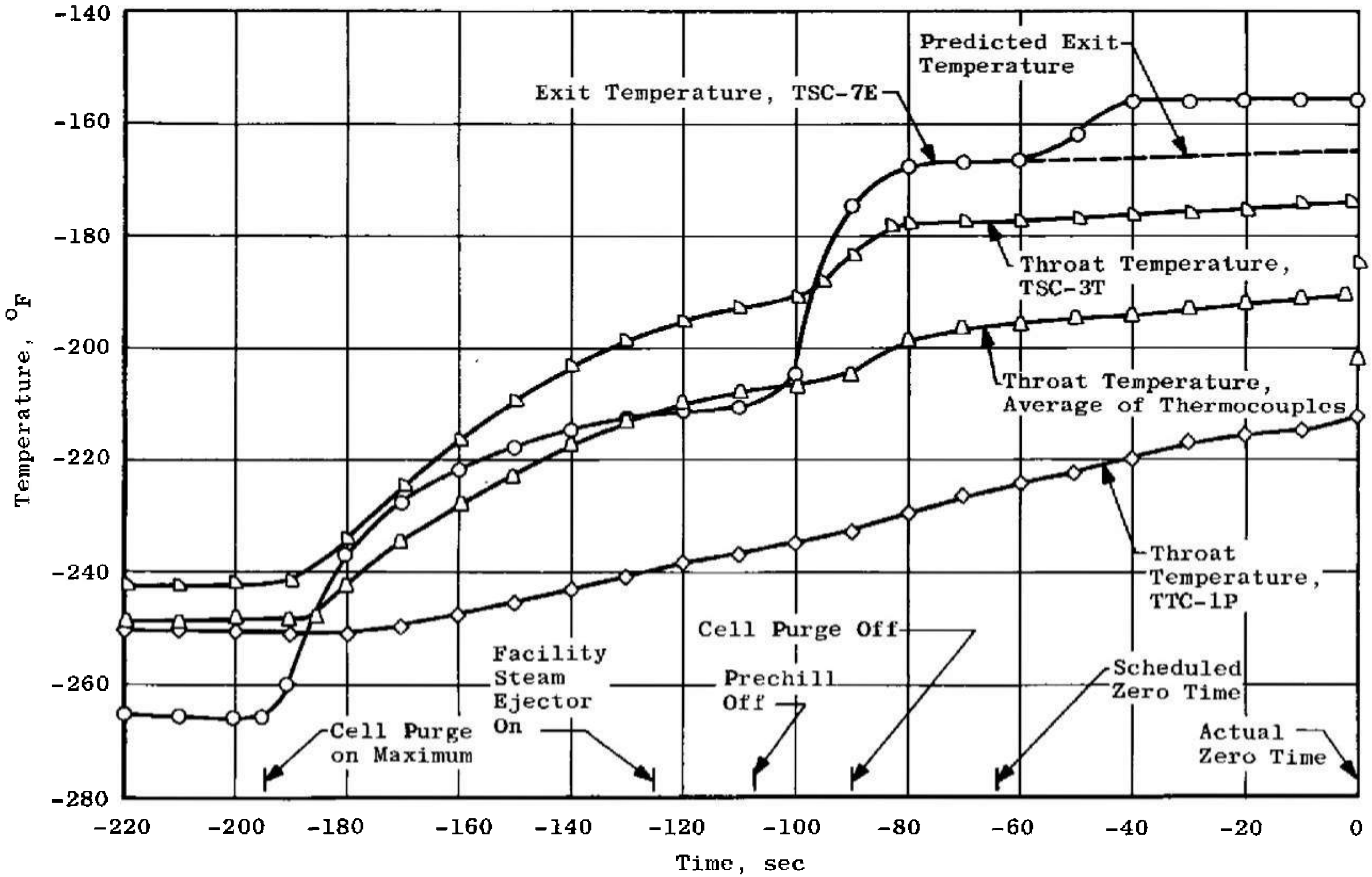


Fig. 31 Pre-Fire Warmup Trends of the J-2 Engine Thrust Chamber

TABLE I
LIST OF ENGINE MODIFICATIONS AT AEDC

Mod. Number	Completion Date	Description of Change
ECP-213	8 July 1966	Replacement of STDV Assembly
ECP-227	8 July 1966	Deletion of Altitude Oxidizer Turbine Bypass Orifice from Engine Loose Equipment and Removal of Ground Test Orifice from Engine
RFD 1-66	29 June 1966	Installation of Fuel Turbine and ASI Propellant Thermocouples
ECP-230	8 July 1966	Replacement of ASI Oxidizer Supply Tube Assembly
RFD 2-66	20 July 1966	Changes to Gas Generator Overtemperature Panel – Low Temperature Timer
ECP-214	9 August 1966	Replacement of MOV Assembly
RFD 3-66	20 September 1966	Installation of Oxidizer Dome Purge Diffuser
RFD 4-66	28 September 1966	Reset Vibration Safety Cutoff Set Delay Timer
ECP-219	15 August 1966	Installation of Filter Assembly to Pneumatic Accumulator – Primary Flight Instrumentation Package
RFD 5-66	14 October 1966	Replaced Oxidizer Dome Purge Diffuser
ECP-235	9 November 1966	Removal of Drain and Instrumentation Bosses from Thrust Chamber Exhaust Gas Manifold Assembly

ECP - Engineering Change Proposal
RFD - Rocketdyne Field Directive

TABLE II
LIST OF ENGINE COMPONENT REPLACEMENTS

Mod. Number	Completion Date	Item Replaced
UCR-004189	3 August 1966	Oxidizer Dome Accelerometer
UCR-004193	19 August 1966	Oxidizer Dome Purge Check Valve
UCR-004187	30 August 1966	Main Oxidizer Flow Pickup
UCR-004183	6 September 1966	GGOT Transducer
UCR-004191	6 September 1966	MOV
UCR-004185	8 September 1966	He Accumulator Hose Assembly
UCR-004182	10 October 1966	Oxidizer Dome Accelerometer
UCR-004117	14 October 1966	Fuel Injector Temperature Transducer

UCR - Unsatisfactory Condition Report

TABLE III
INSTRUMENTATION DESIGNATION OF SELECTED PARAMETERS

Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart
	Temperatures				°F				
1	Fuel Pump Inlet	TFPI-1	C637		-425 to -415	x			
2	Fuel Pump Inlet	TFPI-2	C658		-425 to -400	x	x		x
3	Oxidizer Pump Inlet	TOPI-1	C638		-300 to -275	x			
4	Oxidizer Pump Inlet	TOPI-2	C659		-310 to -270	x	x		x
5	Fuel Pump Discharge	TFPD-1P	C134	PFT1	-425 to -400	x	x	x	
6	Fuel Pump Discharge	TFPD-2	C644	PFT1	-425 to -400	x			
7	Oxidizer Pump Discharge	TOPD-1P	C133	POT3	-300 to -250	x		x	x
8	Oxidizer Pump Discharge	TOPD-2	C648	POT3	-300 to -250	x			
9	Main Fuel Injection	TFJ-1P	C200	CFT2	-425 to +100	x		x	
10	ASI Fuel Injection	TFASIJ		IFT1	-425 to +100	x		x	
11	ASI Oxidizer Injection	TOASIJ		IOT1	-300 to +100	x		x	
12	Fuel Turbine Inlet	TFTI-1P	C001	TGT1	0 to 1800	x			x
13	GG Overtemperature	TGGO-1A	C755	GGT1	0 to 2000	x		x	
14	Fuel Turbine Outlet	TFTO	C654	TGT2	0 to 1800	x			
15	Oxidizer Turbine Inlet	TOTI-1P	C002	TGT3	0 to 1200	x			
16	Oxidizer Turbine Outlet	TOTO-1P	C215	TGT4	0 to 1000	x			
17	Fuel Start Tank	TFST-1P	C006	TFT1	-350 to +100	x			

TABLE III (Continued)

Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart
	Temperatures				°F				
18	Fuel Start Tank	TFST-2	C649	TFT1	-350 to +100				
19	He Tank	THET-1P	C007	NNT1	-350 to +100	x			
20	Electrical Controls Package	TECP-1P	C011	NST1A	-300 to +200	x			x
21	Electrical Controls Package	TECP-2	C657	NST1B	-300 to +200	x		x	
22	Primary Instrument Package	TPIP-1P	C197		-300 to +200	x			
23	Auxiliary Instrument Package	TAIP-1A	C198		-300 to +200	x			
24	Thrust Chamber Jacket (Control)	TTC-1P	C199	CS1	-425 to +100	x			x
25	Thrust Chamber Jacket	TTC-2	C645	CS1A	-425 to +100	x			
26	Thrust Chamber Skin	TSC-1T			-300 to +250	x			
27	Thrust Chamber Skin	TSC-1E			-300 to +250	x			
28	Thrust Chamber Skin	TSC-2E			-300 to +250	x			
29	Thrust Chamber Skin	TSC-3T			-300 to +250	x			
30	Thrust Chamber Skin	TSC-3E			-300 to +250	x			
31	Thrust Chamber Skin	TSC-4T			-300 to +250	x			
32	Thrust Chamber Skin	TSC-4E			-300 to +250	x			
33	Thrust Chamber Skin	TSC-5T			-300 to +250	x			
34	Thrust Chamber Skin	TSC-5E			-300 to +250	x			

TABLE III (Continued)

Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart
	Temperatures				°F				
35	Thrust Chamber Skin	TSC-6T			-300 to +250	x			
36	Thrust Chamber Skin	TSC-6E			-300 to +250	x			
37	Thrust Chamber Skin	TSC-7E			-300 to +250	x			x
38	Thrust Chamber Skin	TSC-8T			-300 to +250	x			x
39	Thrust Chamber Skin	TSC-8E			-300 to +250	x			
40	Oxidizer Recirculation Pump Return Line	TORPR	C159		-300 to -140	x			
41	Fuel Recirculation Pump Return Line	TFRPR	C161		-425 to -250	x			
42	Oxidizer Recirculation Pump Outlet	TORPO	C163		-300 to -250	x			
43	Fuel Recirculation Pump Outlet	TFRPO	C157		-425 to -410	x			
44	Fuel Tank	TFRT-1	C506		-425 to -410	x			
45	Fuel Tank	TFRT-2	C507		-425 to -410	x			
46	Oxidizer Tank	TORT-1	C527		-300 to -287	x			
47	Oxidizer Tank	TORT-2	C528		-300 to -287	x			
	Pressures				psia				
48	Fuel Pump Inlet	PFPI-1	D503		100	x			
49	Fuel Pump Inlet	PFPI-2	D536		200	x	x	x	x
50	Oxidizer Pump Inlet	POPI-1	D504		100	x			

TABLE III (Continued)

Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart
	Pressures				psia				
51	Oxidizer Pump Inlet	POPI-2	D537		200	x	x	x	x
52	Fuel Pump Discharge	PFPD-1P	D008	PF3	1500	x			
53	Fuel Pump Discharge	PFPD-2	D516	PF2	1500	x	x	x	
54	Oxidizer Pump Discharge	POPD-1P	D009	P03	1500	x			
55	Oxidizer Pump Discharge	POPD-2	D522	P02	1500	x	x	x	
56	Main Fuel Injection	PFJ-1A	D004	CF2	1000	x		x	
57	Main Fuel Injection	PFJ-2	D518	CF2A	1000	x			
58	Main Oxidizer Injection	POJ-1A	D005	C03	1000	x			
59	Main Oxidizer Injection	POJ-2		C03A	1000	x	x	x	
60	Thrust Chamber	PC-1P	D001	CG1	1000	x			x
61	Thrust Chamber	PC-2	D544	CG1	1000		x	x	
62	Thrust Chamber	PC-3	D524	CG1A	1000	x		x	
63	ASI Chamber	PCASI	D514	IG1	1000	x		x	
64	Fuel Jacket Inlet Manifold	PFMI		CF1	2000	x			
65	GG Fuel Injection	PFJGG-1A	D011	GF4	1000	x			
66	GG Fuel Injection	PFJGG-2	D527	GF4	1000	x		x	
67	GG Oxidizer Injection	POJGG-1A	D012	G05	1000	x		x	

TABLE III (Continued)

Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart
	Pressures				psia.				
68	GG Oxidizer Injection	POJGG-2	D529	G05	1000	x			
69	GG Chamber	PCGG-1P	D010	GG1	1000	x		x	
70	GG Chamber	PCGG-2	D530	GG1A	1000	x			
71	Fuel Turbine Outlet	PFTO		TG2	200	x			
72	Oxidizer Turbine Inlet	POTI-1A	D007	TG3	200	x			
73	Oxidizer Turbine Outlet	POTO-1A	D086	TG4	100	x			
74	Oxidizer Turbine Outlet	POTO-2	D533	TG4	100	x			
75	Fuel Start Tank	PFST-1P	D017	TF1	1500	x		x	
76	Fuel Start Tank	PFST-2	D525		1500				
77	He Tank	PHET-1P	D019	NN1	3500	x		x	
78	He Tank	PHET-2	D534	NN1	3500				
79	Fuel Recirculation Pump Return Line	PFRPR	D062		50	x			
80	Oxidizer Recirculation Pump Return Line	PORPR	D061		100	x			
81	Fuel Recirculation Pump Outlet	PFRPO	D060		60	x			
82	Oxidizer Recirculation Pump Outlet	PORPO	D059		115	x			
83	Fuel Tank Ullage	PFUT	D539		100	x			
84	Oxidizer Tank Ullage	POUT	D540		100	x			

TABLE III (Continued)

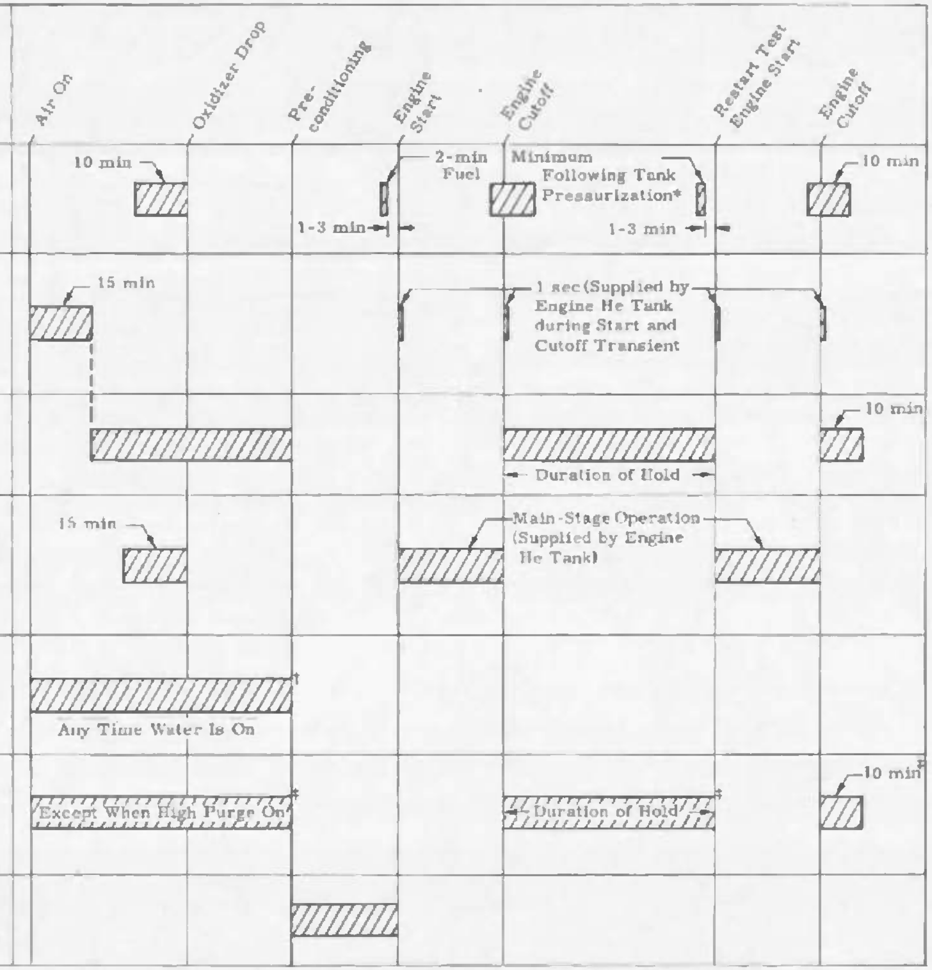
Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillograph	Strip Chart
	<u>Flows</u>				<u>gpm</u>				
85	Fuel Flowmeter	QF-1A	F002	PFF	9000	x		x	
86	Fuel Flowmeter	QF-2	F507	PFFA	9000	x	x	x	x
87	Oxidizer Flowmeter	QO-1A	F001	POF	3000	x		x	
88	Oxidizer Flowmeter	QO-2	F506	POFA	3000	x	x	x	x
89	Fuel Recirculation Flowmeter	QFRP	F005		160	x			
90	Oxidizer Recirculation Flowmeter	QORP	F004		50	x			
	<u>Speeds</u>				<u>rpm</u>				
91	Fuel Pump Speed, rpm	NFP-1P	T002	PFV	30, 000	x	x	x	
92	Oxidizer Pump Speed, rpm	NOP-1P	T001	POV	12, 000	x	x	x	
93	Fuel Chilldown Motor Speed, rpm	NFRP	T501		15, 000	x			
94	Oxidizer Chilldown Motor Speed, rpm	NORP	T500		15, 000	x			
	<u>Vibrations</u>				<u>g</u>				
95	Thrust Chamber Dome	UTCD-1	E706		±150 g		x	x	
96	Thrust Chamber Dome	UTCD-2	E707		±150 g		x	x	
97	Fuel Pump Radial 90 deg	UFPR	E556		±200 g		x		
98	Oxidizer Pump Radial 90 deg	UOPR	E555		±200 g		x		

TABLE III (Concluded)

Item No.	Parameter	AEDC Code	NASA Code	Tap No.	Range	Micro-SADIC	Magnetic Tape	Oscillo-graph	Strip Chart
	Forces				lb_f				
99	Thrust Chamber Pitch	FSP-1	N502		±20,000	x		x	x
100	Thrust Chamber Yaw	FSY-1	N503		±20,000	x		x	x
101	Thrust Chamber Pitch	FSP-2			±20,000		x	x	
102	Thrust Chamber Yaw	FSY-2			±20,000		x	x	
	Pressures				psia				
103	Test Cell	PA1			0.5	x	x	x	
104	Test Cell	PA2			1.0	x	x		
105	Test Cell	PA3			5.0	x			x
	Temperatures				°F				
106	Test Cell	TA1			-50 to +150	x			
107	Test Cell	TA2			-50 to +150	x			
108	Test Cell	TA3			-50 to +150	x			
109	Test Cell	TA4			-50 to +150	x			
	Rates								
110	GG Spark No. 1	RGGS-1	M708		0 to 15 v			x	
111	ASI Spark No. 1	RASIS-1	M706		0 to 15 v			x	
	Events								
112	VSC	U1VSC						x	
113	VSC	U2VSC						x	

TABLE IV
ENGINE PURGE SEQUENCE AT AEDC

Turbopump and GG (Purge Manifold System)	He, 82 - 125 psia 50 - 200°F at Customer Connect Panel, 6 scfm Nominal
Oxidizer Dome and GG Oxidizer Injector (Engine Pneumatic System)	He, 400 ± 25 psig at Engine Pneumatic Package Outlet, 50 to 200°F at He Fill Customer Connect, 230 scfm Nominal
Oxidizer Dome (Facility Line to Instrumentation Port COA3)	N_2, 400 - 450 psig, 100 - 200°F at Facility Check Valve, 200 scfm Minimum
Oxidizer Turbopump Intermediate Seal Cavity (Engine Pneumatic System)	He, 400 ± 25 psig at Engine Pneumatic Package Outlet, Engine Ambient Temperature, 2600 - 7000 scim
Thrust Chamber Jacket (Purge and Preconditioning through Customer Connect Line)	He, 40 - 60 psig, 50 - 200°F at Customer Connect Panel, 60 scfm Nominal
	He, 12 - 14 psig 50 - 200°F at Customer Connect Panel, 10 scfm Nominal
	He, 1000 psig at Customer Connect Panel, 10 - 20 lb_m/min



*Limitation imposed.

†When water is off, purge occurs only for 15 min after oxidizer drop.

‡Except when high purge on.

TABLE V
SUMMARY OF TEST CONDITIONS AND RESULTS

Test Number		02	03	04	05	06	07	08	09A	09B	10A	10B	11A	11B	11C
Test Date		8-27-66	9-2-66	9-14-66	9-24-66	9-30-66	10-7-66	10-18-66	10-27-66	10-27-66	11-4-66	11-4-66	11-16-66	11-16-66	11-16-66
Pressure Altitude at ES, ft		85,600	77,300	87,200	53,300	59,000	73,000	75,300	62,400	92,300	105,700	109,200	109,000	104,000	108,700
Firing Duration, sec		1.169	1.052	1.082	10.075	10.070	20.070	20.078	5.068	0.454	5.070	30.068	40.070	1.378	0.071
Fuel Pump Inlet Conditions at ES	Pressure, psia	37.8	37.2	36.2	36.0	36.2	36.3	35.4	33.1	34.9	35.1	35.8	36.4	36.0	40.3
	Temperature, °F	-420.2	-419.9	-420.1	-419.9	-420.5	-419.8	-420.1	-420.9	-421.3	-420.9	-421.2	-420.2	-420.9	-419.4
Oxidizer Pump Inlet Conditions at ES	Pressure, psia	40.4	39.9	40.1	40.0	40.1	40.6	40.1	41.2	38.8	40.7	38.5	39.2	40.8	39.5
	Temperature, °F	-295.1	-295.6	-295.1	-295.3	-295.0	-295.3	-295.0	-295.4	-289.9	-296.0	-289.2	-296.1	-289.7	-289.8
Start Tank Conditions at ES	Pressure, psia	1,241	1,248	1,265	1,324	1,332	1,323	1,314	1,264	1,350	1,353	1,362	1,362	1,362	N/A
	Temperature, °F	-244	-260	-251	-199	-195	-204	-173	-187	-190	-200	-211	-198	-205	N/A
He Tank Conditions at ES	Pressure, psia	3,006	3,032	3,059	3,005	3,016	3,024	3,008	3,033	3,143	3,243	3,000	3,099	2,990	N/A
	Temperature, °F	-244	-260	-251	-199	-195	-204	-173	-188	-190	-200	-211	-190	-205	N/A
Thrust Chamber Temperature Conditions at ES, °F	Throat	-288	-252	-239	-234	-171	-90	-226	-217	+47	-213	+37	-176	+71	+47
	Exit	-167	-192	-211	-206	-175	-76	-78	-150	+36	-156	+24	-110	+37	34
Fuel Lead Time, sec		1.032	1.051	1.031	1.031	1.050	1.079	1.048	1.046	7.937	1.037	4.584	1.072	8.125	8.243
Fuel in Engine Time, min		104	61	61	108	118	102	60	67	84	71	117	66	94	10
Oxidizer in Engine Time, min		104	61	61	108	118	102	60	67	84	71	117	66	94	10
Propellant Recirculation Time, min		31	11	11	58	10	12	10	12	11	11	5	10	10	10
VSC Duration Time, msec		30	12	132	67	0	30	15	3	N/A	25	0	83	20	N/A
GG Temperature	Initial Spike	1,700	2,230	1,880	1,860	1,740	1,650	1,740	1,700	N/A	1,810	1,580	1,790	2,000	N/A
	Second Spike	---	---	---	1,670	1,680	1,490	1,610	1,730	N/A	1,870	1,680	1,730	2,150	N/A
Thrust Chamber Ignition, sec		1.034	0.997	0.979	0.973	0.970	0.978	0.990	0.904	N/A	0.962	0.973	0.958	0.987	N/A
MOV Initial Opening Time, sec		1.012	1.000	1.000	1.004	1.008	1.110	1.150	1.155	N/A	1.200	1.173	1.212	1.290	N/A
Main-Stage "OK" Signal, sec		N/A	N/A	N/A	1.644	1.646	1.648	1.720	1.685	N/A	1.698	1.623	1.650	N/A	N/A
550-psia Chamber Pressure Attained, sec		N/A	N/A	N/A	1.919	1.917	1.951	1.960	1.910	N/A	1.867	1.925	1.878	N/A	N/A

TABLE VI
ENGINE SITE PERFORMANCE SUMMARY

Test Number			05	06	07		08	10B			11A			
Data Slice Time, sec			9.5	9.5	9.5	19.5	19.5	9.5	19.5	29.5	0.5	10.5	20.5	30.5
Pressure, Altitude, ft			63,800	66,300	83,300	80,350	87,000	100,300	100,300	88,500	08,500	97,300	96,000	87,500
Overall Engine Performance	Thrust, lb_f	Site	190,180	190,630	100,710	102,170	192,930	191,120	182,690	192,220	217,850	220,060	220,130	210,350
		Vacuum	194,370	194,240	192,320	194,020	194,290	191,870	193,420	193,610	218,650	220,910	221,020	220,670
	Specific Impulse, $\frac{lb_f\text{-sec}}{lb_m}$	Site	420.4	418.4	424.1	424.2	422.0	423.1	428.6	424.1	421.8	422.2	422.7	422.2
		Vacuum	429.5	426.3	427.7	428.3	425.0	424.7	430.2	427.2	423.4	423.8	424.4	424.8
	Mixture Ratio		4.860	4.878	4.845	4.888	4.922	4.832	4.830	4.855	5.404	5.492	5.518	5.448
	Fuel Weight Flow, lb_m/sec		77.19	77.50	76.92	76.93	77.20	77.48	77.12	77.42	80.85	80.28	79.92	80.57
	Oxidizer Weight Flow, lb_m/sec		375.2	378.1	372.7	376.0	379.9	374.28	372.5	375.8	435.8	440.9	440.8	438.9
	Total Weight Flow, lb_m/sec		452.4	455.6	449.6	453.0	457.1	451.74	449.6	453.3	516.5	521.2	520.8	519.5
Thrust Chamber Performance	Specific Impulse, lb_f-sec/lb_m		426.3	424.1	430.1	430.2	427.9	429.0	434.6	430.0	427.5	427.9	428.4	427.9
	Characteristic Velocity, ft/sec		8,099	8,038	8,071	8,079	8,015	8,015	8,118	8,057	7,925	7,928	7,939	7,946
	Thrust Coefficient	Site	1.694	1.698	1.714	1.713	1.718	1.722	1.723	1.717	1.736	1.736	1.736	1.732
		Vacuum	1.730	1.730	1.729	1.730	1.730	1.729	1.730	1.730	1.742	1.743	1.743	1.743
	Mixture Ratio		5.045	5.061	5.029	5.073	5.109	5.015	5.013	5.038	5.613	5.704	5.730	5.658
	Fuel Weight Flow, lb_m/sec		73.80	74.25	73.55	73.55	73.80	74.07	73.73	74.03	77.06	76.72	76.36	76.98
	Oxidizer Weight Flow, lb_m/sec		372.3	375.3	360.9	373.2	377.0	371.4	369.6	373.0	432.6	437.8	437.5	435.6
	Total Weight Flow, lb_m/sec		446.1	449.4	443.4	446.7	450.8	445.5	443.4	447.0	509.6	514.3	513.9	512.6
	Chamber Pressure, psia		659.2	659.1	652.9	658.4	659.2	651.5	656.5	657.1	736.8	744.0	744.3	743.2
	Fuel Injector	Stagnation Pressure, psia	749.8	748.4	741.1	746.4	748.1	742.5	746.6	747.4	835.4	842.3	844.4	844.5
		Temperature, °F	-295.0	-292.4	-205.3	-295.0	-297.8	-298.5	-287.2	-287.5	-278.0	-276.8	-278.3	-274.5
	Oxidizer Injector	Stagnation Pressure, psia	791.3	784.0	775.2	789.6	794.0	786.0	788.7	796.8	903.6	921.3	925.6	926.9
Fuel Pump Performance	Inlet	Pressure, psia	34.4	34.7	34.4	34.6	33.8	34.2	34.5	34.7	34.8	35.3	34.8	34.4
		Density, lb_m/ft^3	4.32	4.35	4.32	4.32	4.35	4.36	4.36	4.35	4.32	4.32	4.32	4.32
	Discharge	Pressure, psia	1,072	1,071	1,059	1,064	1,068	1,059	1,065	1,066	1,188	1,192	1,192	1,194
		Density, lb_m/ft^3	4.41	4.41	4.41	4.41	4.48	4.46	4.46	4.46	4.45	4.44	4.44	4.44
	Head, ft		33,280	33,020	32,880	33,040	32,970	32,630	32,820	32,880	36,800	37,000	37,020	37,090
	Speed, rpm		25,660	25,480	25,220	25,210	25,570	25,310	25,360	25,350	26,710	26,750	26,760	26,770
	Weight Flow, lb_m/sec		77.19	77.50	76.92	76.93	77.20	77.46	77.12	77.42	80.65	80.28	79.92	80.57
	Volume Flow, gpm		8,010	7,980	7,986	7,987	7,962	7,980	7,946	7,979	8,376	8,339	8,303	8,372
	Efficiency		0.704	0.684	0.711	0.711	0.702	0.733	0.733	0.732	0.732	0.730	0.729	0.728

TABLE VI (Concluded)

Test Number			05	06	07		08	10B			11A			
Data Slice Time, sec			9.5	9.5	9.5	19.5	19.5	9.5	19.5	20.5	0.5	19.5	20.5	30.5
Oxidizer Pump Performance	Inlet	Stagnation Pressure, psia	40.1	39.7	40.0	39.8	39.9	39.5	39.4	39.3	38.5	38.7	38.4	38.8
		Density, lb_m/ft^3	70.95	70.86	70.81	70.79	70.89	69.82	69.81	69.80	71.10	71.09	71.08	71.05
	Discharge	Stagnation Pressure, psia	890.9	889.8	878.0	886.0	893.5	882.2	881.2	893.0	1,041	1,051	1,052	1,051
		Density, lb_m/ft^3	70.96	70.87	70.82	70.79	70.88	69.83	69.81	69.80	71.31	71.29	71.28	71.25
	Head, ft		1,727	1,727	1,704	1,721	1,734	1,738	1,757	1,761	2,024	2,045	2,047	2,045
	Speed, rpm		7,795	7,783	7,741	7,796	7,819	7,811	7,868	7,875	8,370	8,435	8,453	8,451
	Weight Flow, lb_m/sec		455.4	458.6	451.0	455.9	461.0	452.4	452.1	455.8	443.4	440.5	448.4	446.5
	Volume Flow, gpm		2,881	2,905	2,859	2,890	2,910	2,908	2,907	2,931	2,799	2,832	2,832	2,820
	Efficiency		0.810	0.811	0.810	0.810	0.811	0.810	0.810	0.810	0.799	0.800	0.800	0.799
	PU Valve Position, deg		-0.7	-0.8	-0.8	-0.8	-0.7	-0.7	-0.7	-0.7	31.3	31.3	31.3	31.3
GG Performance	Chamber Pressure (Measured), psia		582.0	577.7	575.6	582.5	588.8	577.0	581.8	582.0	648.2	653.4	653.3	656.5
	Mixture Ratio		0.848	0.852	0.839	0.850	0.854	0.838	0.847	0.849	0.911	0.928	0.930	0.929
	Fuel Weight Flow, lb_m/sec		3.39	3.35	3.37	3.38	3.40	3.39	3.39	3.38	3.58	3.57	3.56	3.58
	Oxidizer Weight Flow, lb_m/sec		2.87	2.85	2.83	2.88	2.90	2.83	2.87	2.87	3.27	3.31	3.31	3.33
	Total Weight Flow, lb_m/sec		6.26	6.20	6.21	6.26	6.30	6.23	6.26	6.25	6.85	6.88	6.87	6.91
Fuel Turbine Performance	Weight Flow, lb_m/sec		6.26	6.20	6.21	6.26	6.30	6.23	6.26	6.25	6.85	6.88	6.87	6.91
	Inlet	Static Pressure, psia	541.1	537.1	535.3	541.8	545.6	535.4	541.0	541.1	603.7	608.7	608.5	611.8
		Temperature, °F	1,044	1,052	1,029	1,048	1,059	1,024	1,043	1,047	1,152	1,180	1,183	1,182
	Exit	Static Pressure, psia	74.81	74.35	73.44	74.91	75.35	73.77	75.27	75.53	82.23	84.14	84.37	84.39
		Temperature, °F	606.2	606.2	589.1	636.6	645.4	588.8	633.9	543.8	671.5	734.4	746.6	747.3
	Pressure Ratio		7.42	7.41	7.48	7.42	7.43	7.46	7.38	7.35	7.53	7.42	7.40	7.45
	Developed Horsepower		6,632	6,805	6,460	6,503	6,594	6,267	6,280	6,320	7,375	7,387	7,378	7,459
	Efficiency		0.603	0.623	0.595	0.591	0.593	0.576	0.573	0.577	0.587	0.584	0.583	0.585
Oxidizer Turbine Performance	Weight Flow, lb_m/sec		5.57	5.52	5.53	5.58	5.61	5.54	5.57	5.57	6.10	6.13	6.13	6.17
	Inlet	Static Pressure, psia	72.5	72.1	71.1	72.5	72.9	71.5	72.9	73.2	79.6	81.5	81.7	81.8
		Temperature, °F	606.2	606.2	589.1	636.8	645.4	588.8	633.9	643.8	671.5	734.4	746.6	747.3
	Exit	Static Pressure, psia	27.63	27.39	27.13	27.93	28.09	27.22	28.00	28.17	30.61	31.67	31.93	31.87
		Temperature, °F	437.5	436.8	419.7	483.9	503.6	423.3	481.9	506.7	477.8	576.4	595.1	598.4
	Pressure Ratio		2.67	2.67	2.67	2.84	2.84	2.87	2.85	2.64	2.64	2.62	2.60	2.60
	Developed Horsepower		1,765	1,777	1,725	1,701	1,793	1,764	1,783	1,801	2,041	2,085	2,087	2,078
	Efficiency		0.456	0.464	0.455	0.446	0.449	0.463	0.452	0.454	0.473	0.464	0.463	0.457

TABLE VII
S-IVB STAGE PERFORMANCE SUMMARY

Test Number		02	03	04	05	06	07	08	09A	09B	10A	10B	11A	11B	11C
Stage Propellant Conditions	Fuel Ullage Pressure, psia	19.7	19.7	19.1	21.3	21.2	20.8	19.6	17.4	16.4	16.9	16.6	19.0	17.1	17.8
	Fuel Bulk Temperature, °F	-421.2	-421.2	-421.5	-420.7	-420.7	-420.8	-421.2	-422.0	-422.4	-422.2	-422.3	-421.4	-422.1	-421.8
	Oxidizer Ullage Pressure, psia	14.7	14.9	14.6	15.5	17.1	18.1	15.7	14.2	23.2	14.3	21.9	14.3	21.9	20.3
	Oxidizer Bulk Temperature, °F	-296.4	-296.6	-296.7	-296.7	-295.3	-295.9	-296.3	-296.9	-291.1	-297.0	-290.6	-297.6	-291.1	-291.5
Fuel Recirculation System	Pump Flow, gpm	128.4	126.3	122.9	129.6	134.3	130.9	129.4	130.8	131.5	128.1	133.3	128.5	133.1	112.0
	Pump Speed, rpm	11,610	11,600	11,620	11,610	11,600	11,610	11,600	11,590	11,600	11,610	11,610	11,600	11,600	11,640
	Pump Outlet Pressure, psia	28.5	28.7	28.6	30.4	29.5	29.0	28.0	25.9	25.0	26.0	25.5	27.8	25.9	27.2
	Pump Outlet Temperature, °F	---	---	---	---	---	---	---	---	---	---	---	-421.0	-421.8	-421.4
	Pump Return Pressure, psia	20.3	20.6	20.1	22.0	22.0	21.5	20.5	18.3	17.4	18.0	17.7	19.9	18.0	18.8
	Pump Return Temperature, °F	---	---	---	---	---	---	---	---	---	---	---	-419.1	-419.3	-419.1
Oxidizer Recirculation System	Pump Flow, gpm	36.4	35.9	36.2	36.5	37.6	38.4	---	35.8	38.6	36.0	37.3	36.3	36.9	35.6
	Pump Speed, rpm	11,750	11,720	11,740	11,770	11,750	11,750	11,770	11,760	11,710	11,750	11,750	11,760	11,750	11,780
	Pump Outlet Pressure, psia	24.6	24.3	24.4	24.4	26.6	26.7	25.0	26.8	37.0	26.6	35.1	26.5	33.2	32.1
	Pump Outlet Temperature, °F	-297.7	-297.8	-297.8	-297.7	-297.6	-297.3	-297.7	-297.9	-293.3	-296.9	-290.4	-297.4	-290.8	-291.2
	Pump Return Pressure, psia	17.7	17.4	16.6	17.6	19.3	20.4	18.3	16.9	25.6	16.8	24.2	16.8	23.9	21.3
	Pump Return Temperature, °F	-294.6	-294.8	-294.6	-294.9	-294.6	-294.5	-294.8	-295.3	-289.2	-295.2	-288.6	-295.6	-289.3	-290.5

Measurement obtained before pressurization of propellant tanks.

TABLE VIII
AEDC TEXT MATRIX

Test Number		02	03	04	05	06	07	08	09A	09B	10A	10B	11A	11B	11C
Fuel Pump Inlet Conditions at ES	Pressure, psia	35.5 ± 1.5	35.5 ± 1.5	35.5 ± 1.5	35.5 ± 1.5	35.5 ± 1.5	35.5 ± 1.5	35.5 ± 1.5	33 ± 0.5	37^{+0}_{-3}	33 ± 0.5	37^{+0}_{-3}	34^{+3}_{-0}	34^{+3}_{-0}	-
	Temperature, °F	-419.9 ± 0.4	-419.8 ± 0.4	-419.9 ± 0.4	-419.9 ± 0.4	-419.9 ± 0.4	-420 ± 0.4	-419.9 ± 0.4	-420.8 ± 0.4	-421.1 ± 0.4	-420.8 ± 0.4	-421.2 ± 0.4	-419.9 ± 0.4	-421.1 ± 0.4	-
Oxidizer Pump Inlet Conditions at ES	Pressure, psia	39.5 ± 1.5	39.5 ± 1.5	39.5 ± 1.5	39.5 ± 1.5	39.5 ± 1.5	39.5 ± 1.5	39.5 ± 1.5	41 ± 0.5	38^{+3}_{-0}	41 ± 0.5	38^{+3}_{-0}	41^{+0}_{-3}	38^{+3}_{-0}	-
	Temperature, °F	-295.1 ± 0.4	-295.1 ± 0.4	-295.1 ± 0.4	-295.1 ± 0.4	-295.1 ± 0.4	-295.2 ± 0.4	-295.1 ± 0.4	-295.6 ± 0.4	-290 ± 0.4	-295.7 ± 0.4	-290.1 ± 0.4	-296.4 ± 0.4	-289.9 ± 0.4	-
Start Tank Conditions at ES	Pressure, psia	1350 ± 25	1250 ± 25	1250 ± 25	1250 ± 25	1250 ± 25	1350 ± 25	1325 ± 25	1250 ± 25	1335 ± 25	1350 ± 25	1335 ± 25	1375 ± 25	1350 ± 25	-
	Temperature, °F	-250.0 ± 25	-250.0 ± 25	-250.0 ± 25	-250.0 ± 25	-250.0 ± 25	-250.0 ± 25	-170 ± 10	-200 ± 25	-200 ± 20	-200 ± 25	-200 ± 25	-200 ± 25	-200^{+25}_{-0}	-
Minimum Fuel Lead Time, sec		1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	8.0	1.0	6.5	1.0	8.0	8.0
Fuel in Engine, min		120	120	120	60	60	60	60	60	†	60	†	60	†	10
Oxidizer in Engine, min		120 ± 20	120 ± 20	120 ± 20	60	60	60	60	60	†	60	†	60	†	10
Propellant Recirculation Time, min	Fuel	10	10	10	10	10	10	10	10	10	10	5	10	10	10
	Oxidizer	10	10	10	10	10	10	10	10	10	10	5	10	10	10
Thrust Chamber Temperature Conditions at ES, °F	Throat	**	**	-225 ± 25	-275 ± 25	-150 ± 25	-100 ± 15	-220 ± 10	-220 ± 10	N/A	-220 ± 10	N/A	-165 ± 10	N/A	N/A
	Exit	N/A	N/A	N/A	N/A	N/A	N/A	-170 ± 10	-170 ± 10	N/A	-170 ± 10	N/A	-125 ± 10	N/A	N/A

**Temperature condition to coldest temperature attainable before simulated boost phase warmup period.
†Propellants on engine for duration of coast period before firing.

TABLE IX
START TRANSIENT OPERATING TIMES OF SELECTED ENGINE VALVES

Firing Number	Start																						
	STDV, Opening			Main Fuel Valve, Opening			MOV, First Stage, Opening			MOV, Second Stage, Opening			GG Fuel Poppet, Opening			GG Oxidizer Poppet, Opening			Oxidizer Turbine Bypass Valve, Closing			STDV, Closing		
	Time of Opening Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec	Time of Opening Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec	Time of Closing Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Closing Time, sec	Time of Closing Signal, Reference to STDV Solenoid "On"	Valve* Delay Time, sec	Valve Opening Time, sec
02	0	0.152	0.145	-1.097	0.041	0.061	0.435	0.058	0.047	0.435	N/A	N/A	0.536	0.080	0.045	0.536	0.177	0.070	0.536	0.100	0.274	0.435	0.089	0.273
03	0	0.160	0.140	-1.061	0.045	0.062	0.441	0.095	0.000	0.141	N/A	N/A	0.546	0.077	0.053	0.546	0.179	0.078	0.546	0.100	0.276	0.441	0.100	0.256
04	0	0.144	0.136	-1.031	0.042	0.045	0.442	0.050	0.048	0.442	N/A	N/A	0.540	0.071	0.057	0.540	0.171	0.067	0.540	0.100	0.352	0.442	0.100	0.257
05	0	0.156	0.112	-1.031	0.045	0.062	0.452	0.051	0.060	0.152	0.570	2.110	0.553	0.079	0.061	0.553	0.175	0.067	0.553	0.101	0.275	0.452	0.100	0.270
06	0	0.178	0.147	-1.080	0.047	0.066	0.441	0.052	0.053	0.441	0.533	2.067	0.546	0.060	0.050	0.546	0.172	0.065	0.546	0.089	0.270	0.441	0.091	0.275
07	0	0.160	0.154	-1.079	0.050	0.070	0.448	0.055	0.050	0.440	0.575	2.000	0.545	0.074	0.054	0.545	0.164	0.070	0.545	0.112	0.256	0.440	0.100	0.270
08	0	0.142	0.140	-1.045	0.049	0.066	0.442	0.050	0.050	0.442	0.801	2.012	0.542	0.073	0.056	0.542	0.170	0.070	0.542	0.111	0.262	0.442	0.097	0.266
09A	0	0.144	0.137	-1.046	0.047	0.070	0.442	0.051	0.058	0.442	0.714	1.988	0.543	0.073	0.051	0.543	0.171	0.070	0.543	0.109	0.255	0.442	0.095	0.281
09B	0	0.151	0.149	-1.047	0.060	0.059	0.442	N/A	N/A	0.442	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	0.442	N/A	N/A
10A	0	0.152	0.143	-1.037	0.050	0.062	0.440	0.052	0.048	0.440	0.710	1.942	0.540	0.073	0.053	0.540	0.148	0.011	0.540	0.117	0.256	0.440	0.089	0.250
10B	0	0.158	0.157	-1.584	0.051	0.065	0.439	0.055	0.055	0.438	0.730	2.075	0.549	0.063	0.053	0.549	0.185	0.078	0.549	0.100	0.271	0.438	0.086	0.259
11A	0	0.150	0.145	-1.072	0.042	0.072	0.440	0.048	0.052	0.440	0.654	1.881	0.540	0.078	0.064	0.540	0.177	0.072	0.540	0.104	0.262	0.440	0.091	0.260
11B	0	0.145	0.154	-1.125	0.050	0.066	0.443	0.049	0.058	0.443	0.738	N/A	0.550	0.080	0.063	0.550	0.176	0.074	0.550	0.092	0.269	0.443	0.094	0.274
11C	0	N/A	N/A	-1.343	0.049	0.063	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A

* Valve delay time is the time required for initial valve movement after the valve "open" or "closed" solenoid has been energized.

N/A Not Applicable, firing terminated before valve attained full open or closed position.

TABLE X
SHUTDOWN TRANSIENT OPERATING TIMES OF SELECTED ENGINE VALVES

Firing Number	Shutdown									
	Main Fuel Valve, Closing		MOV, Closing		GG Fuel Poppet, Closing		GG Oxidizer Poppet, Closing		Oxidizer Turbine Bypass Valve, Opening	
	Valve Delay* Time, sec	Valve Closing Time, sec	Valve Delay* Time, sec	Valve Closing Time, sec	Valve Delay* Time, sec	Valve Closing Time, sec	Valve Delay* Time, sec	Valve Closing Time, sec	Valve Delay* Time, sec	Valve Opening Time, sec
02	0.117	0.307	0.037	0.043	0.067	0.021	0.111	0.010	0.129	0.760
03	0.114	0.297	0.032	0.036	0.065	0.018	0.104	0.014	0.129	0.673
04	0.103	0.277	0.037	0.034	0.080	0.022	0.147	0.022	0.139	0.677
05	0.121	0.327	0.075	0.180	0.072	0.020	0.117	0.007	0.222	0.560
06	0.126	0.322	0.077	0.182	0.085	0.020	0.140	0.010	0.217	0.571
07	0.150	0.316	0.079	0.140	0.081	0.015	0.129	0.012	0.245	0.580
08	0.120	0.300	0.075	0.167	0.083	0.020	0.141	0.010	0.238	0.616
09A	0.122	0.309	0.054	0.177	0.081	0.021	0.136	0.012	0.222	0.590
09B	0.110	0.309	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A
10A	0.118	0.303	0.057	0.174	0.080	0.017	0.133	0.014	0.231	0.612
10B	0.118	0.328	0.084	0.188	0.078	0.017	0.127	0.015	0.242	0.579
11A	0.113	0.292	0.076	0.177	0.080	0.027	0.138	0.014	0.254	0.593
11B	0.110	0.325	0.076	0.188	0.070	0.025	0.136	0.018	0.165	0.533
11C	0.097	0.275	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A

*Valve delay time is the time required for initial valve movement after the engine cutoff command has been initiated.

N/A Not Applicable; firing terminated before valve attained full open or closed position.

TABLE XI
TEST MEASUREMENTS REQUIRED BY PERFORMANCE PROGRAM

Item No.	Parameter
1	Thrust Chamber (Injector Face) Pressure, psia
2	Thrust Chamber Fuel and Oxidizer Injection Pressures, psia
3	Thrust Chamber Fuel Injection Temperature, °F
4	Fuel and Oxidizer Flowmeter Speeds, cps
5	Fuel and Oxidizer Engine Inlet Pressures, psia
6	Fuel and Oxidizer Pump Discharge Pressures, psia
7	Fuel and Oxidizer Engine Inlet Temperatures, °F
8	Fuel and Oxidizer (Main Valves) Temperatures, °F
9	PU Valve Center Tap Voltage, v
10	PU Valve Position, v
11	Fuel and Oxidizer Pump Speeds, rpm
12	GG Chamber Pressure, psia
13	GG (Bootstrap Line at Bleed Valve) Temperature, °F
14	Fuel* and Oxidizer Turbine Inlet Pressure, psia
15	Oxidizer Turbine Discharge Pressure, psia
16	Fuel and Oxidizer Turbine Inlet Temperature, °F
17	Oxidizer Turbine Discharge Temperature, °F

*At AEDC, fuel turbine inlet pressure is estimated from gas generator chamber pressure.

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13. ABSTRACT

Eleven test periods involving a total of 14 J-2 rocket engine starts were accomplished in Propulsion Engine Test Cell (J-4) to verify the engine altitude ignition characteristics and performance in support of the Saturn IB and Saturn V flights. The test article consisted of a flight configuration J-2 rocket engine and a "battleship" S-IVB stage. Engine ignition characteristics were different from those predicted for altitude testing and provided an important insight into the J-2 engine transient operation for the Saturn IB and Saturn V flights. Transient gas generator outlet temperatures observed were consistently higher than engine acceptance test values. One test resulted in an engine safety cutoff because of an excessive transient gas generator outlet temperature. Combustion instability during engine ignition was observed for time durations ranging from 3 to 132 msec for ten of the firings and was similar to the instability recorded during the Saturn IB AS-201 through -203 flights. Excessive engine vibration resulted in an engine safety cutoff on one test. Engine side forces measured at altitude conditions were significantly less than those recorded during previous J-2 engine tests. Engine operation time accumulated during these tests was 145.8 sec.

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